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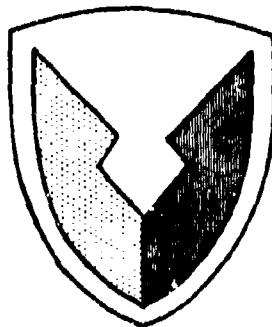
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## AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST

### PRODUCTION OH-58A HELICOPTER UNARMED AND ARMED WITH XM27EI ARMAMENT SUBSYSTEM

#### STABILITY AND CONTROL

#### FINAL REPORT

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OCTOBER 1970

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US ARMY AVIATION SYSTEMS TEST ACTIVITY  
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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## **ABSTRACT**

Stability and control tests were conducted on a production model OH-58A helicopter to evaluate its flying qualities in the unarmed configuration and in an armed configuration with the XM27E1 weapon system. Limited testing was also performed to evaluate the helicopter slope landing capabilities and flying qualities with skis installed. Human factors and maintainability characteristics were noted throughout the evaluation. Testing was performed by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California, between 6 October 1969 and 16 February 1970. The testing consisted of 89 flights which totaled 85.3 hours of productive flight testing. There were no deficiencies recorded. Eighteen shortcomings are reported. Difficulty in maintaining precise directional control during hovering flight is a shortcoming which warrants improvement on a priority basis. This shortcoming requires excessive pilot effort and degrades the accuracy in firing of the XM27E1 armament subsystem. It is recommended that a caution note be placed in the operator's manual warning against hovering in a tail wind in excess of 30 knots. The capability of landing on a 10-degree slope was marginal but is not considered to be a shortcoming. Flying qualities of the aircraft with skis installed were satisfactory. Maintainability of the helicopter was excellent throughout the test program.

## **FOREWORD**

Throughout the OH-58A stability and control testing, technical support was provided under contract by the airframe manufacturer, Bell Helicopter Company, Fort Worth, Texas, and the engine manufacturer, Allison Division of General Motors Corporation, Indianapolis, Indiana. Instrument calibration, emergency fire fighting, scientific photography and medical support were provided by the US Air Force Flight Test Center, Edwards Air Force Base, California.

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# INTRODUCTION

## BACKGROUND

1. A limited engineering flight test of the Bell Model 206A helicopter (JetRanger) was conducted during the US Army Light Observation Helicopter (LOH) procurement competition from September through December 1967. The Bell Helicopter Company (BHC) of Fort Worth, Texas, was subsequently awarded a production contract to build a modified version of the Model 206A with the military designation OH-58A.
2. During the period from 26 June 1969 through 9 July 1969, the Army Preliminary Evaluation (APE) of the OH-58A was conducted at the BHC facility. This APE consisted of limited quantitative and qualitative stability and control testing with the OH-58A in the armed scout configuration, as defined in the detail specification (ref 1, app 1). Thirteen test flights were performed, totaling 9.1 productive hours.
3. On 7 August 1968, the US Army Aviation Test Activity (USAATTA), now the US Army Aviation Systems Test Activity (USAASTA), was directed (ref 2, app 1) by the US Army Aviation Systems Command (USAASCOM) to conduct Airworthiness and Flight Characteristics (A&FC) testing on the OH-58A helicopter. The testing was divided into two phases: Performance, and Stability and Control, with separate reports required for each phase. Performance testing was completed in January 1970, and the final report was published in 1970 (ref 3). This report contains the final results of the stability and control testing.

## TEST OBJECTIVES

4. The objectives of the OH-58A stability and control tests were to determine the capability of the helicopter to perform its intended mission and to verify compliance with the requirements of the military specification (mil spec) MIL-H-8501A as amended by deviation 19 of the detail specification (ref 4, app 1). Special tests were conducted to evaluate the aircraft's handling qualities during firing of the XM27E1 armament subsystem and under various conditions with skis fastened to the skid landing gear. A qualitative analysis was performed to determine the slope-landing capability of the OH-58A, maintenance characteristics and human factors relating to the helicopter. Specification compliance was determined both with the XM27E1 armament subsystem installed and removed.

DESCRIPTION

5. The OH-58A light observation helicopter has a single main rotor, and an antitorque tail rotor which are two-bladed, semirigid and teetering. The tail rotor also has a delta three hinge. The cockpit provides side-by-side seating for a crew of two (pilot and copilot/observer), and the cargo compartment has seats for two passengers. Dual flight controls are provided. The cyclic and collective controls are hydraulically boosted and irreversible, while the antitorque tailrotor control is unboosted. The main landing gear consists of fixed, energy-absorbing skids. The helicopter is powered by an Allison T63-A-700 free gas turbine engine with a takeoff power rating of 317 shaft horsepower (shp) under sea-level (SL), standard-day uninstalled conditions. The main transmission has a rating of 270 shp for continuous operation with a takeoff power limit of 317 shp (5-minute rating). More detailed aircraft information may be found in reference 5, appendix I.

6. The XM27E1 armament subsystem consists of one XM134 high rate 7.62 millimeter (mm) (GAU-2B/A) with mount, feed system and ammunition boxes, and one XM70E1 weapon sight. The armament subsystem is mounted on the left side of the helicopter near the longitudinal center of gravity (cg). The XM134 gun is adjustable in elevation from 5 degrees above to 20 degrees below waterline (WL) zero and is operated by either the pilot or copilot/observer. It will fire at either 2000 or 4000 rounds per minute (rds/min). The ammunition capacity is 2000 rounds.

SCOPE OF TEST

7. Stability and control tests on the OH-58A were conducted at forward flight speeds ranging from 30 to 129 knots calibrated airspeed (KCAS). Hover and sideward and rearward flight testing was also performed. Approximate gross weights (grwt) ranged from 2245 to 2990 pounds at density altitudes from sea level to 15,000 feet. The longitudinal center of gravity (cg) was varied from full forward to full aft.

8. Testing was conducted in the armed configuration and with the XM27E1 armament subsystem removed. The flying qualities of the helicopter with the pilot door and the two passenger doors removed were evaluated and compared with the doors-on configuration.

9. Testing was conducted in California at Bishop (elevation 4112 feet), Coyote Flats (elevation 9500 feet), Shafter (elevation 420 feet), and Edwards Air Force Base (elevation 2302 feet). The test program was conducted from October 1969 to February 1970 and consisted of 89 flights totaling 85.3 productive flight hours.

METHOD OF TEST

10. The test methods utilized are outlined in the test plan (ref 6, app I) and are discussed further in the Results and Discussion section of this report. All tests were conducted under nonturbulent atmospheric conditions to preclude uncontrolled disturbances from influencing the test data.

11. An OH-58A helicopter (S/N 68-16706) was equipped with sensitive calibrated instruments. A detailed list of the recorded parameters is presented in appendix II. The pilot's comments were used to aid in the analysis of data and to assist in the overall qualitative assessment of the flying qualities of the OH-58A. The Handling Qualities Rating Scale (HQRS) is included as appendix III.

CHRONOLOGY

12. The chronology of this test:

Test directive issued	7 August	1968
Test plan published	May	1969
Test helicopter received	7 August	1969
First stability and control test flight	6 October	1969
A&FC testing completed	16 February	1970
Draft report submitted	June	1970

## RESULTS AND DISCUSSION

### STATIC LONGITUDINAL STABILITY

13. Static longitudinal collective-fixed stability and static trim characteristics tests were performed. The static longitudinal stability of the helicopter was evaluated in level flight and climb by varying the airspeed in increments of approximately 7 knots using the longitudinal cyclic control while maintaining the collective fixed at the trim position. Control positions were recorded while the helicopter was stabilized at each incremental airspeed above and below the trim airspeed. Summary plots of the collective-fixed curves are presented in figure 1, appendix IV, and show the effects of changes in density altitude ( $H_D$ ), gross weight (gr wt), helicopter configuration, flight condition and cg. Detailed data plots showing the results of the collective-fixed tests are presented in figures 2 through 9. The level-flight conditions are listed in table 1. Collective-fixed tests were also conducted in climbs.

Table 1. Static Longitudinal Collective-Fixed Stability Flight Test Conditions.

Configuration	Density Altitude (ft)	Gross Weight (lb)	Center of Gravity (in.)
Armed <sup>1</sup>	880	2,660	105.8 (fwd)
Armed <sup>1</sup>	6,360	2,660	106.7 (fwd)
Armed <sup>1</sup>	15,030	2,665	106.7 (fwd)
Armed <sup>1</sup>	5,960	2,900	106.1 (fwd)
Armed <sup>1</sup>	6,290	2,950	112.1 (aft)
Armed <sup>2</sup>	6,090	2,680	106.4 (fwd)
Clean <sup>1</sup>	5,990	2,620	106.2 (fwd)
Clean <sup>1</sup>	6,040	2,340	111.0 (mid)

<sup>1</sup>Doors on.

<sup>2</sup>Doors off.

14. Static longitudinal trim characteristics were investigated by measuring control displacement at trim conditions in level flight, autorotation and climb. Airspeed was varied in increments of approximately 10 knots by changing power. These data are summarized in figure 10, appendix IV, and presented in detail in figures 11 through 18. Flight conditions and helicopter configurations for the tests were approximately the same as those listed in table 1.

#### Collective-Fixed Characteristics

15. The longitudinal control position gradient with airspeed was stable (negative) for all flight conditions during the collective-fixed testing although the position gradient was weak (shallow slope) for the 2340-pound grwt, mid cg condition at a level-flight trim speed of 59 KCAS (fig. 1, app IV). A shallow position gradient results in a lack of control displacement cue as airspeed changes; however, in this case, it was not of such a reduced magnitude as to be objectionable to the pilot (HQRS 3). Since a linear relationship exists between the longitudinal control force and the control position when the force trim is turned ON, the control force gradient was also negative for all force-trim ON flight conditions.

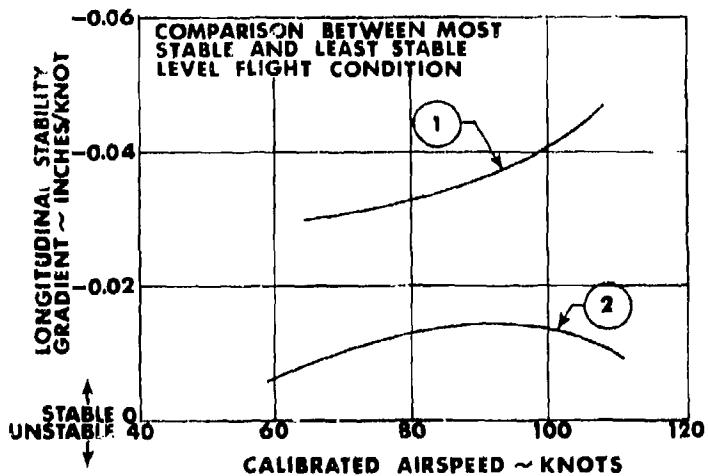
16. No clearly defined trend was exhibited by the effects of density altitude on the collective-fixed static longitudinal stability. At the lowest test altitude (880-ft  $H_D$ ), as shown in figure 1, appendix IV, there was a more negative position gradient than for the 6360-foot  $H_D$ . At airspeeds above 65 KCAS, however, the highest test density altitude (15,030 ft) also resulted in a gradient which reflected more static stability than at the 6360-foot  $H_D$  condition.

17. The effects of center of gravity on the collective-fixed static longitudinal stability in level flight were well defined. The stability was less with aft cg locations. The helicopter exhibited more negative gradients with increasing gross weights. Removal of the doors had no noticeable affect on static longitudinal stability. Removal of the weapons system, however, resulted in a more negative gradient (fig. 1, app IV). The combined effects of changing the individual parameters discussed above are shown in figure A. In this figure, the least negative stability gradient (2340-lb grwt and mid cg) is compared with the most stable configuration tested (2620-lb grwt and fwd cg).

18. For several flight conditions, as shown in figure 1, appendix IV, the longitudinal stability gradient for climb was greater than that for level flight. For all flight conditions, the climb gradient was at least equal to the level-flight gradient.

**FIGURE A**  
**COLLECTIVE FIXED LONGITUDINAL STABILITY**

CURVE NUMBER	CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT	CONFIGURATION
1	106.2 (FWD)	2620	5990	CLEAN (DOORS ON)
2	111.0 (MID)	2340	6040	CLEAN (DOORS ON)



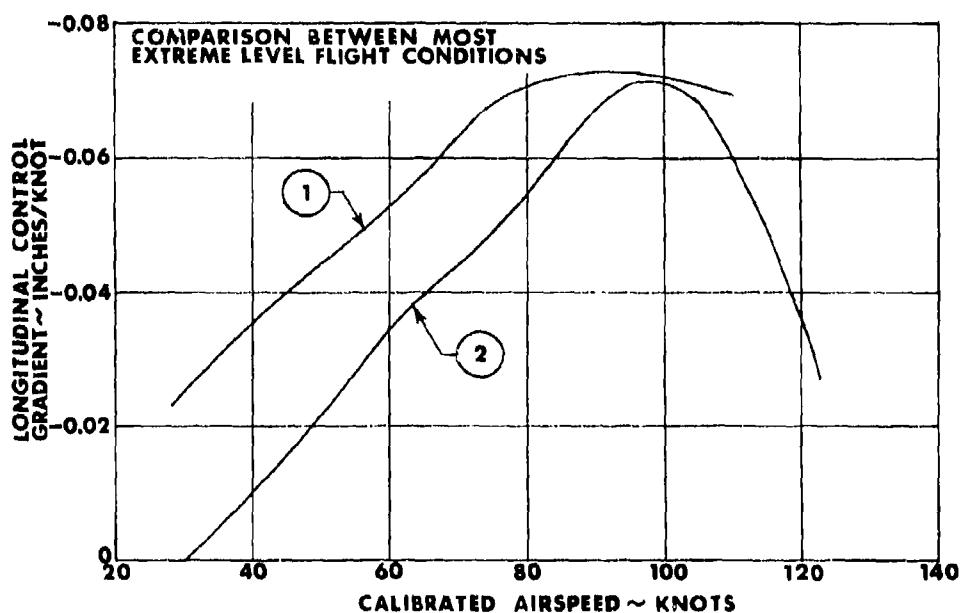
Static Longitudinal Trim Characteristics

19. Figure 10, appendix IV, is a summary plot which shows the results of the tests that determined the control positions in forward level flight, climb and autorotation. A comparison of the weakest control position gradient with the most negative gradient is shown in figure B.

20. In the static trim characteristics tests, as in the collective-fixed test, the control position gradients became less negative as the cg was moved aft or gross weight was decreased (fig. 10, app IV). The weakest gradient for each configuration was evident in autorotation. The difference between climb and level flight was negligible. Varying the helicopter configuration from the clean to the armed configuration or by removing the doors had a negligible effect upon the static longitudinal trim characteristics. For all configurations tested, the longitudinal control position gradient approached zero between 20 and 30 KCAS. Positive gradients were found to exist at lower forward speeds (see discussion on low-speed forward flight under the heading: Sideward and Rearward Flight, paragraph 58). All control margins were adequate and met the requirement of the mil spec at the forward cg; however, at two conditions (figs. 15 and 17), at a mid and an aft cg, the mil spec forward longitudinal control margin requirement of 10 percent remaining was reached at 120 KCAS. The data indicate that this control margin could not be achieved at light grwt, full aft cg conditions.

**FIGURE B**  
**LONGITUDINAL CONTROL GRADIENTS**

CURVE NUMBER	CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT	CONFIGURATION
1	105.9(FWD)	2695	6040	CLEAN (DOORS ON)
2	111.1(MID)	2360	6030	CLEAN (DOORS ON)



21. At the highest airspeed tested (approximately 90 KCAS), autorotations required 2 to 3 inches more aft cyclic stick displacement than did climbs. The longitudinal cyclic deviation between autorotation and climb exceeded the 3-inch limit (para 3.2.10.2 of the mil spec for three test conditions at 90 KCAS (figs. 12, 17 and 18, app IV). Since the helicopter configuration, cg, and gross weight were different for each of the three instances, no consistent trend could be defined. It was noted that for several other conditions tested, the longitudinal cyclic deviation between climb and autorotation approached the limit of the mil spec at the highest airspeed tested. Qualitatively, however, this deviation was not objectionable to the pilot during flight.

22. There was generally less than 1 inch of lateral cyclic variation between climb and autorotation. Pedal displacement between climb and autorotation varied from 1 to 2 inches. Variations in the lateral cyclic movement and pedal travel with changes in airspeed were insignificant. The mil spec requirements (para 3.3.17) for the lateral cyclic travel were satisfied.

23. A few minor problems were encountered during the static longitudinal stability testing. It was difficult to maintain precise rotor rpm control during autorotation, *i.e.*, small variations in pitch attitude or sideslip angle would cause large fluctuations in rotor speed ( $\pm 20$  rpm). This condition should not pose a problem for normal mission accomplishment. Control of rotor speed in autorotation was considerably easier at a 15,000-foot  $H_D$  than at 6000 feet. Also, it was easier to stabilize on level-flight points at the higher altitude. During the low-speed testing (below 40 knots indicated airspeed (KIAS)), stabilized flight was relatively difficult to maintain because of the apparent influence of the aircraft long-period mode and a slight Dutch-roll oscillation which occurred occasionally (discussed more fully under the heading: Static Lateral-Directional Stability).

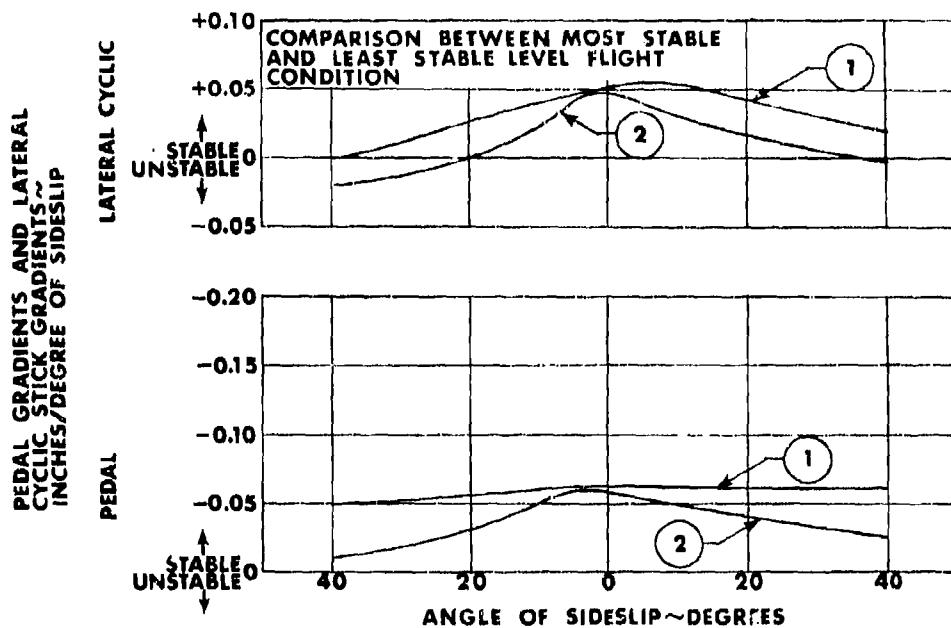
24. For all flight conditions, cyclic control forces could be easily reduced to zero by the use of the force trim system. "Stick jump" (an unwanted control motion resulting from pressing the force trim button while a force is being held against the cyclic control) was negligible during the flight testing. There was no evidence of cross-coupling between the cyclic and the collective controls during flight with the boost ON. Notwithstanding the mil spec noncompliances mentioned in paragraph 21, the static longitudinal collective-fixed and trim characteristics of the helicopter were satisfactory for all conditions tested (HQRS 2).

#### STATIC LATERAL-DIRECTIONAL STABILITY

25. Static lateral-directional stability was tested under the flight conditions listed in table 2. The tests were conducted by establishing a trim airspeed with zero sideslip and varying the sideslip angles while maintaining constant airspeed and ground track. The boom-mounted, pitot-static swivel probe used during the stability and control testing, eliminated sideslip-induced errors in the indicated airspeed. Control positions were recorded for each steady-heading sideslip. The results of the static lateral-directional stability tests are summarized in figure 19, appendix IV, and presented in detail in figures 20 through 29. A comparison between the most stable (2670-lb grwt, fwd cg) and the least stable (2360-lb grwt, mid cg) lateral and directional control position gradients is presented in figure C.

**FIGURE C**  
**STATIC LATERAL DIRECTIONAL STABILITY GRADIENTS**

CURVE NUMBER	CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT	CALIBRATED AIRSPEED ~KTS
1	107.2(FWD)	2670	14920	53
2	111.1(MID)	2360	6120	53



26. The directional control pedal position versus sideslip gradient was stable (negative) for all level flight and climb flight conditions and aircraft configurations (fig. 19, app IV). The pedal position gradients decreased with decreasing gross weights and aft cg movement. Pedal position gradients increased with increasing airspeed for all flight conditions. Density altitude variations showed no clearly defined trend. The effect of other test condition variables was negligible (*ie*, doors and armament subsystem configurations) upon the level-flight, static lateral-directional stability.

27. Bank-angle gradients were essentially linear which indicates that there were positive side force characteristics for all level-flight and climb conditions tested. Side force increased with increasing airspeeds and provided strong cues as to the directional stability of the helicopter.

Table 2. Static Lateral-Directional Test Conditions.

Configuration	Altitude (ft)	Gross Weight (lb)	Center of Gravity (lb)	Flight Mode
Armed <sup>1</sup>	1,660	2,735	105.9 (fwd)	Level
Armed <sup>1</sup>	5,980	2,570	106.9 (fwd)	Level
Armed <sup>1</sup>	5,790	2,620	107.0 (fwd)	Climb, autorotation
Armed <sup>1</sup>	14,920	2,670	107.2 (fwd)	Level
Armed <sup>1</sup>	6,020	2,840	105.6 (fwd)	Level
Armed <sup>1</sup>	5,930	2,960	112.1 (aft)	Level
Armed <sup>2</sup>	5,990	2,655	106.5 (fwd)	Level
Clean <sup>1</sup>	5,970	2,650	105.7 (fwd)	Level
Clean <sup>1</sup>	6,120	2,360	111.1 (mid)	Level

<sup>1</sup>Doors on.

<sup>2</sup>Doors off.

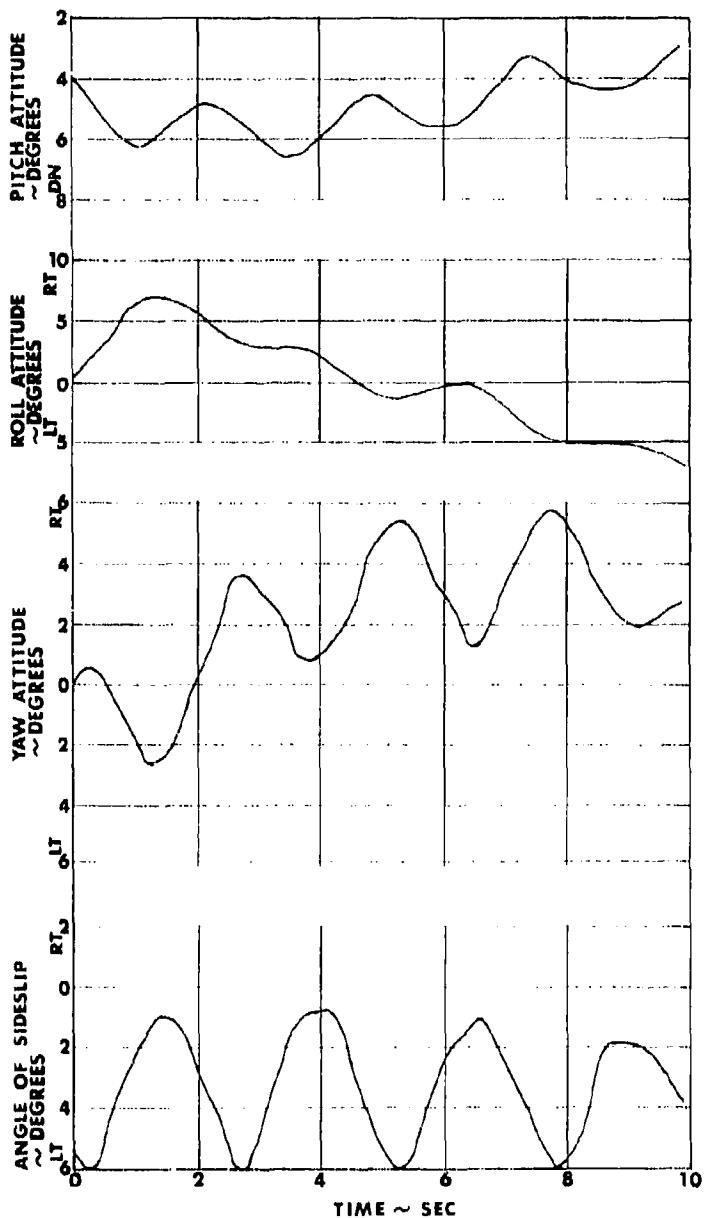
28. Lateral cyclic control position gradients approached zero at 53 KCAS in level flight and 49 KCAS in climb at sideslip angles greater than 30 degrees both right and left (fig. 21, app IV). Although effective dihedral was weak at the high-sideslip angle flight conditions, it was not an objectionable characteristic for normal, operational flying at low-sideslip angles. The effective dihedral was found to increase with increasing gross weights and increasing airspeeds. The effect of varying the cg was negligible (fig. 19).

29. In autorotation at 49 KCAS, the pedal-position gradient (fig. 22, app IV) was essentially neutral at the zero-sideslip point. For the same flight condition, the lateral stick-position gradient and the dihedral effect were unstable (negative) for sideslip angles greater than 25 degrees left, slightly stable (positive) between 25 degrees left and 30 degrees right, and neutral above a 30-degree right sideslip. In autorotation at 85 KCAS, a reversal (from positive to negative) in the slope of the lateral control occurred at sideslip angles of approximately 15 degrees both right and left (fig. 23). Qualitatively, this gradient reversal would not be objectionable to the pilot during normal mission flying. Bank-angle gradients found during autorotation indicated the presence of a linear side force characteristic which increased in magnitude with airspeed. The operational pilot could fly the helicopter inadvertently in a small degree of sideslip in either direction, while performing a low airspeed autorotation, because of the combination of a weak positive dihedral effect and a weak pedal-position gradient (HQRS 3). This condition should not adversely affect mission accomplishment.

30. During the static lateral-directional stability testing, a significant Dutch-roll oscillation was encountered at left sideslip angles of approximately 5 degrees. Typical time histories of the oscillation are shown in figure D and also in figure 30, appendix IV. The oscillation was moderately damped at lower airspeeds, but the damping effect decreased as airspeed was increased. At the higher airspeeds the Dutch-roll oscillation was neutrally damped. Damping increased when the doors were removed and when the armament subsystem was not installed. The oscillation is not likely to occur during the normal operations (when the helicopter is in a zero-sideslip or slight right-sideslip condition) except during very turbulent conditions, but would be extremely objectionable during firing if the pilot inadvertently placed the helicopter in a slight left sideslip (HQRS 4). In such a situation the firing accuracy would be significantly degraded. Correction of this shortcoming is desirable.

**FIGURE D**  
**DUTCH ROLL**

CALIBRATED AIRSPEED ~KTS	CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT	CONFIGURATION
93	105.7(FWD)	2740	2950	ARMED(DOORS ON)



31. At right-sideslip angles greater than 15 degrees with airspeed at 80 KCAS or greater, a high-frequency (21.5 Hertz) vibration was evident in the helicopter. The source of the vibration was observed to be a small amplitude, lateral vibration about the longitudinal axis of the upper vertical stabilizer. A photographic record was made of this vibration. The condition would normally not be encountered during an operational situation where sideslip angles seldom reach this magnitude and is not considered a problem.

32. In general, overall static lateral-directional flying qualities of the OH-58A were acceptable (HQRS 4). The requirements of paragraph 3.3.9 of the mil spec were not met in autorotation at 85 KCAS for the lateral stick and pedal gradients (fig. 23, app IV), in that the gradients were not approximately linear. Also, some gradients were slightly unstable (figs. 22, 23, and 29) at various airspeeds. These characteristics were not objectionable in flight; however, the previously discussed shortcoming accounted for the relatively low pilot rating of the OH-58A in static lateral-directional stability.

#### DYNAMIC STABILITY

33. Longitudinal, lateral, and directional dynamic stability characteristics of the helicopter were tested at approximately the same conditions as listed in table 3. Representative results are presented as time histories in figures 31 through 36, appendix IV.

34. The longitudinal, long-period dynamic stability characteristics were evaluated by stabilizing the helicopter at trim airspeed and then increasing and decreasing the airspeed by desired increments using only the cyclic control. The controls were then returned to the trim position, and the helicopter response was observed by the pilot and recorded on an oscillograph. The long-period oscillatory mode was convergent for all level-flight conditions, the damping increased with increasing airspeed. In a maximum-power climb, the long-period mode was divergent at 35 and 49 KCAS, and damped at 80 KCAS. The period of the oscillation was approximately 20 seconds for all flight conditions. Damping was well within the limits of paragraph 3.2.11 of the mil spec. There was no control coupling present, nor was rotor overspeed a problem. The long-period mode was easily excited at low airspeeds (below 40 KIAS) during level-flight tests (para 23); however, this characteristic should not be detrimental to mission accomplishment.

35. The longitudinal gust response characteristics of the helicopter were tested by applying cyclic pulse inputs to excite the aircraft short-period mode. The test results are presented in figures 31 and 32, appendix IV. For all conditions tested, the short-period mode was heavily damped, and the requirements of paragraph 3.2.11 of the mil spec were met. The longitudinal dynamic stability of the helicopter was satisfactory (HQRS 2).

36. Lateral gust response characteristics were tested by applying lateral pulse inputs to the cyclic control (figs. 33 and 34, app IV). The spiral stability characteristics (ability of the helicopter to return to level flight after a disturbance in the roll axis) indicated a neutral mode. This was most evident during turbulent flight conditions where constant, small lateral control corrections were necessary. Neutral spiral stability was acceptable for the tasks tested (HQRS 3), although it would be objectionable if the helicopter were to be flown under instrument flight conditions.

37. Pedal pulse inputs resulted in lightly to moderately damped yaw oscillations (figs. 35 and 36, app IV). Yaw damping varied directly with airspeed and varied inversely with altitude. At a 3,000-foot  $H_g$ , the yaw axis damping ratio, determined by the transient peak ratio method, was 0.15 at 63 KCAS and increased to 0.30 at 108 KCAS. A directional disturbance usually resulted in a slight nose-down pitching of the helicopter. This characteristic was acceptable for the tasks being evaluated. A lightly damped, Dutch-roll oscillation was occasionally generated by the pedal pulses. This oscillation was also encountered during the static, lateral-directional testing as discussed in paragraph 30. The dynamic lateral and directional flying qualities were satisfactory for the tasks tested (HQRS 3).

#### CONTROL RESPONSE AND SENSITIVITY

##### General

38. The longitudinal, lateral and directional control response (maximum angular rate per inch of control input) and sensitivity (maximum angular acceleration per inch of input) of the helicopter were tested at the flight conditions listed in table 3. The step input method was used for these tests. Each control was rapidly displaced and then held firmly against a rigid fixture until the maximum rate was reached or until recovery became necessary. Resultant attitudes, rates and accelerations were recorded on an oscilloscope. The results are presented in figures 37 through 58, appendix IV. The helicopter responded in the proper direction

In all axes complying with the requirements of paragraphs 3.3.16 and 3.2.9 of the mil spec. The control power (helicopter angular displacement per inch of control input per second) for all axes met the minimum requirements of the mil spec.

Table 3. Control Response and Sensitivity Test Conditions.

Configuration	Density Altitude (ft)	Gross Weight (lb)	Center of Gravity (in.)	Flight Condition
Armed <sup>1</sup>	-30	2,655	106.1 (fwd)	Hover OGE <sup>2</sup>
Armed <sup>1</sup>	3,910	2,685	106.2 (fwd)	Hover OGE
Armed <sup>1</sup>	10,530	2,510	109.1 (mid)	Hover OGE
Armed <sup>1,3</sup>	270	2,960	106.0 (fwd)	Hover OGE
Armed <sup>1,3</sup>	-930	2,810	111.9 (aft)	Hover OGE
Armed <sup>1</sup>	5,960	2,700	107.0 (fwd)	LF <sup>4</sup> , C <sup>5</sup> , A <sup>6</sup>
Armed <sup>1</sup>	6,320	2,865	105.6 (fwd)	LF, C <sup>3</sup> , C <sup>3</sup>
Armed <sup>1</sup>	6,140	2,830	112.0 (aft)	LF, C <sup>3</sup> , A <sup>3</sup>
Armed <sup>1</sup>	14,900	2,660	106.9 (fwd)	LF
Armed <sup>1,3</sup>	3,150	2,680	106.1 (fwd)	LF
Clean <sup>1,3</sup>	5,720	2,675	105.2 (fwd)	LF, C, A
Armed <sup>3,7</sup>	5,950	2,620	106.4 (fwd)	LF, C, A

<sup>1</sup>Doors on.

<sup>4</sup>Level flight.

<sup>2</sup>Out of ground effect.

<sup>5</sup>Climb.

<sup>3</sup>This condition was tested,

<sup>6</sup>Autorotation.

but the data are not presented. <sup>7</sup>Doors off.

#### Longitudinal

39. The maximum longitudinal control response could not be measured in hover. Recovery inputs were required prior to reaching the maximum pitch rate in order to avoid extreme aircraft attitudes. Therefore, longitudinal control response in hover was measured 1 second after the step input was initiated. The pitch rate decreased as airspeed increased and ranged from 10 deg/sec/in. (up and down) in hover at a 3910-foot  $H_D$  to 5.5 deg/sec/in. (up) and 4.5 deg/sec/in.

(down) at a 5960-foot  $H_D$  and 115 KCAS. The effects of different gross weights, cg's, density altitudes and flight conditions (climb and autorotation) were negligible (figs. 37 and 38, app IV). The pitch-axis rate damping met the requirements of paragraph 3.2.14 of the mil spec. An  $I_{yy}$  of 1947 slugs/ft<sup>2</sup>, supplied by Bell Helicopter Company, was used to determine this specification compliance.

40. Airspeed significantly affected the downward pitching sensitivity which ranged from 15 deg/sec<sup>2</sup>/in. in hover to 10 deg/sec<sup>2</sup>/in. at 115 KCAS. The upward pitching acceleration remained essentially constant. Moving the location of the cg had no noticeable effect on the longitudinal sensitivity. The effects of different density altitudes and flight conditions were also negligible. Increasing the gross weight from 2690 to 2980 pounds increased the sensitivity by approximately 2 deg/sec<sup>2</sup>/in. The longitudinal control response and sensitivity were satisfactory (HQRS 2).

#### Lateral

41. The maximum lateral control response in hover was 24 deg/sec/in. at a minus 30-foot  $H_D$  and exceeded the 20-deg/sec/in. limitation of paragraph 3.3.15 of the mil spec (figs. 39 and 40, app IV). This limitation was exceeded at several other flight conditions but was not considered to be objectionable. At a 3910-foot  $H_D$ , it was 20 deg/sec/in. (both right and left). Lateral control response decreased at 49 KCAS to approximately 14 deg/sec/in. (both right and left) but increased with additional increases in airspeed and measured 22 deg/sec/in. (right) at 115 KCAS. In forward flight, the effects of different helicopter configurations, altitudes, gross weights and cg's were insignificant. In a hover, however, the lateral response decreased as density altitude increased. Changing the flight condition at a constant airspeed resulted in the most significant variation in lateral control response. At 49 KCAS, the rate varied from 10 deg/sec/in. (both right and left) in autorotation to 20 deg/sec/in. (both right and left) in climb. Level-flight response at 49 KCAS was 13 to 14 deg/sec/in. These variations are assumed to have resulted from the changes in rotor thrust. The control response increased as thrust increased. At greater airspeeds, the lateral response also varied more between right and left step inputs. The differences between the inputs ranged from zero in hover to 4 deg/sec/in. at 115 KCAS. Although the lateral rate damping did not meet the minimum requirement of paragraph 3.3.19 of the mil spec, it did comply with deviation 19 of the detail specification (ref 1, app I).

42. The lateral sensitivity increased slightly with increasing airspeed and ranged from 30 deg/sec<sup>2</sup>/in. (left) in hover to 34 deg/sec<sup>2</sup>/in. (both right and left) at 115 KCAS. At 49 KCAS, the acceleration varied from 20 deg/sec<sup>2</sup>/in. (both right and left) in

autorotation to 32 deg/sec<sup>2</sup>/in. (right) in climbs. Level-flight sensitivity at 49 KCAS was 26 deg/sec<sup>2</sup>/in. (right) and 24 deg/sec<sup>2</sup>/in. (left). This nonlinear variation with airspeed (which is similar to the lateral response variation) was not objectionable to the pilot. Moving the cg aft increased the lateral sensitivity by approximately 4 deg/sec<sup>2</sup>/in. Gross weight, density altitude, and helicopter configuration effects were negligible. The lateral control response and sensitivity are considered to be satisfactory (HQRS 2).

#### Directional

43. The maximum directional control response in hover could not be measured since the rate increased steadily with time, indicating a lack of yaw rate damping (figs. 57 and 58, app IV). This characteristic is an undesirable shortcoming (HQRS 5), and correction is recommended. Yaw rate should reach a maximum value quickly, dependent only upon the magnitude of the control displacement and not upon the duration of the input. As a result of the lack of yaw rate damping, the time to reach a maximum yaw rate could not be measured as aircraft recovery was necessary before the maximum rate was attained. Using the "time constant" method of determining first order rate damping,

where: Damping =  $\frac{\text{Moment of Inertia}}{\text{Time to } 0.63 \text{ Max Rate}}$

the rate damping approached zero since time to reach the maximum rate was excessive. Therefore, the minimum yaw rate damping requirement of paragraph 3.3.19 of the mil spec was not met. An  $I_{zz}$  of 1534 slugs/ft<sup>2</sup> supplied by Bell Helicopter Company was used to determine this specification noncompliance. The response in hover was measured 1 second after the step input. The maximum directional control response measured occurred in hovering flight and reached a value of 49 deg/sec/in. for the right pedal step input at a 30-foot  $H_D$  (figs. 41 and 42, app IV). The minimum directional control response was 11.0 deg/sec/in. to the right and occurred during forward flight at 66 KCAS and at a 14,900-foot  $H_D$ . Airspeed was the most significant variable which affected directional control response and sensitivity at airspeeds of less than 38 KCAS. Both rates and accelerations increased as airspeed decreased. The maximum control sensitivity was 68 deg/sec<sup>2</sup>/in. (both right and left) which also occurred in hover at the same flight conditions as the maximum control response. The minimum sensitivity was 33.5 deg/sec<sup>2</sup>/in. (both right and left) and occurred at the same flight conditions as the minimum response described above.

44. In level flight, the rates and accelerations decreased significantly as density altitude increased. The effects of different

gross weights, cg's or helicopter configurations were insignificant, and there was no appreciable variation between level flight, climb and autorotation.

45. Longitudinal and lateral couplings were evident during directional-control response testing. A slight downward pitching motion and a roll in the direction of the yaw resulted from the pedal step inputs. This coupling would not adversely affect mission accomplishment.

#### MANEUVERING STABILITY

46. The maneuvering stability characteristics of the helicopter were tested for the flight conditions listed in table 4. The symmetrical pull-up was the primary test technique used during the testing. For this test, the helicopter was trimmed in level flight at the desired airspeed after which a cyclic pull-up to a slightly higher altitude was initiated. A dive was then established, and the helicopter was accelerated to near trim airspeed. A symmetrical pull-up was executed so as to pass through the required trim airspeed, altitude, pitch attitude and the desired load factor, simultaneously. Longitudinal stick force and normal acceleration were recorded at each test point. The results are summarized in figure 59, appendix IV, and presented in detail in figures 60 through 62. The maneuvering stability characteristics were spot-checked during turns at the 2700-pound grwt and forward cg. The results are compared with corresponding data obtained in pull-up maneuvers shown in figure 60.

Table 4. Maneuvering Stability Flight Conditions.

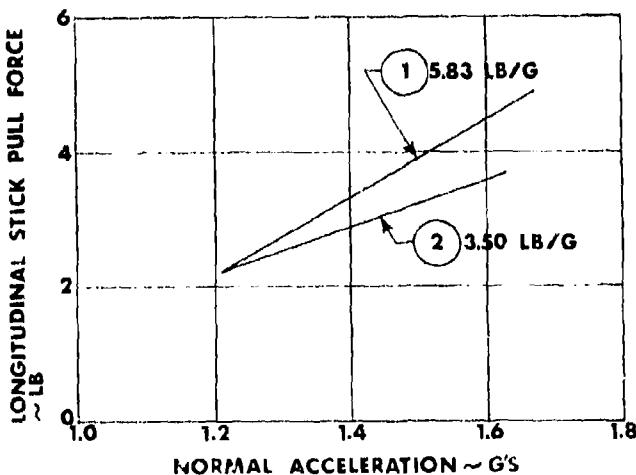
Configuration	Density Altitude (ft)	Gross Weight (lb)	Center of Gravity (in.)	Hydraulic Boost System
Armed <sup>1</sup>	5,230	2,705	107.1 (fwd)	On
Armed <sup>1</sup>	10,950	2,675	107.0 (fwd)	On
Armed <sup>1</sup>	4,950	2,990	106.2 (fwd)	On
Armed <sup>1</sup>	5,350	2,970	112.2 (aft)	On
Armed <sup>1</sup>	5,390	2,630	106.9 (fwd)	Off
Armed <sup>2</sup>	5,080	2,680	106.4 (fwd)	On
Clean <sup>1</sup>	5,200	2,650	106.0 (fwd)	On

<sup>1</sup>Doors on. <sup>2</sup>Doors off.

47. The boost-ON stick force gradient relative to the acceleration ( $F_s/g$ ) was positive for all conditions tested, i.e., a greater aft cyclic force was required for an increased load factor. The force gradient depends entirely on the spring effect of the force trim system. The gradients became more positive with increased gross weight (increasing from 4.43 lb/g at 2705 pounds to 5.83 lb/g at 2990 pounds). The gradients also increased as the cg was moved forward (fig. E). Changing the airspeed, altitude or helicopter configuration had no significant effect on the stick force gradients.

**FIGURE E  
MANEUVERING STABILITY**

CURVE NUMBER	CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT
1	106.2(FWD)	2990	4950
2	112.2(AFT)	2970	5350

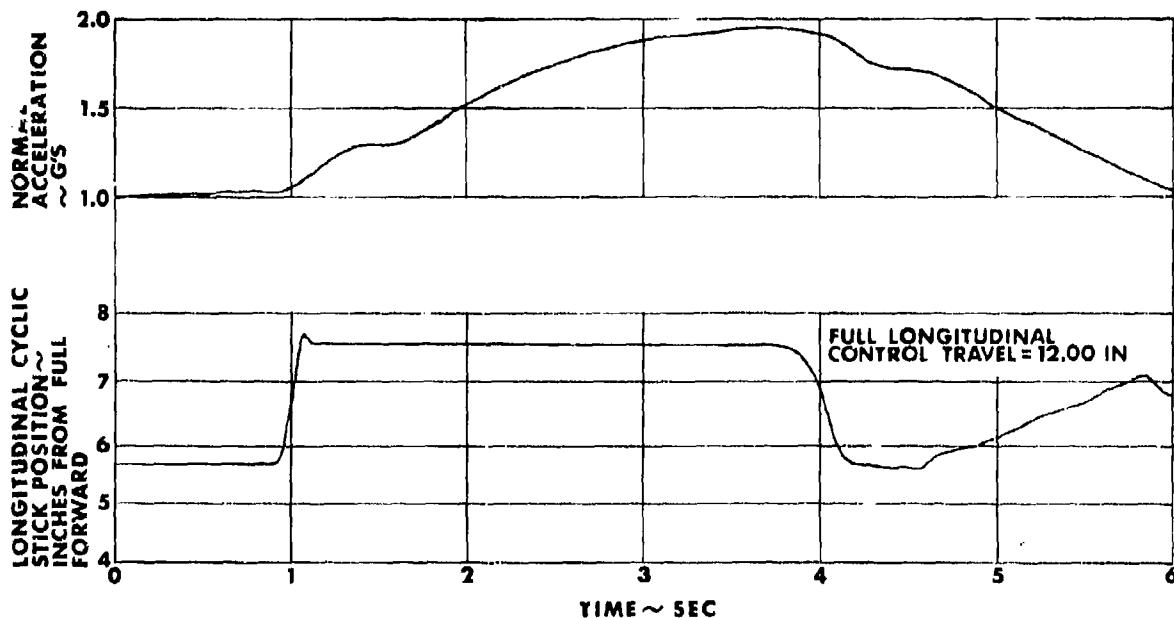


48. The maximum normal acceleration recorded was 1.9 g's at a 2650-pound grwt. It was not possible to reach the limit load factor of 2.8 g's. Although stick-force-per-g gradients are light, it is not likely that the helicopter will be overstressed during normal operation because of the large pitch attitudes required. The helicopter did not tend to roll in either direction during symmetrical pull-ups, nor was any rotor overspeed encountered. At the highest airspeed tested (100 KCAS), blade stall was experienced (as evidenced by a severe 2-per-rev vertical vibration during pull-ups from dives). This vibration was not transmitted through the control system, and adequate warning was provided. The effect was not a problem at the conditions tested.

49. Typical time history plots of normal acceleration during symmetrical pull-ups are shown in figure F and also in figure 61, appendix IV. Although the normal acceleration decayed slightly after an initial rise, it then continued to build and became concave downward within 2 seconds following the start of the maneuver. It then remained concave until maximum acceleration was attained. This characteristic met the requirements of paragraph 3.2.11.1 of the mil spec. The helicopter stick-force-per-g characteristics (boost ON) were satisfactory (HQRS 3).

**FIGURE F  
SYMMETRICAL PULL UP**

CG ~IN	GROSS WEIGHT ~LB	DENSITY ~FT	CALIBRATED AIRSPEED ~KTS
107.0(FWD)	2700	5230	80



50. The maneuvering stability of the helicopter was tested with the boost OFF at a 2650-pound grwt, a 5390-foot H<sub>p</sub> and a forward cg configuration (fig. 62, app IV). Control coupling existed between the longitudinal cyclic stick and the collective pitch, i.e., raising the collective transmitted a pull force to the cyclic while lowering the collective resulted in a cyclic push force. A qualitative evaluation of the test results showed that positive aft cyclic forces were required to obtain positive load factors; however, there was no consistency in the stick force gradient. Because of the excessive

collective and cyclic control forces, the boost-OFF maneuvering characteristics of the helicopter are considered to be satisfactory for emergency operations only (HQRS 6). The correction of this shortcoming is recommended.

#### SIDEWARD AND REARWARD FLIGHT

51. Sideward, rearward and forward flight tests were conducted to determine the hovering capability in ground effect (IGE) of the helicopter in winds of various speeds and azimuths. Azimuths were varied in 30-degree increments measured clockwise from the nose of the helicopter. The test conditions are listed in table 5. Sideward flights were performed at airspeeds up to 35 knots true airspeed (KTAS), and rearward flights were performed at airspeeds up to 30 KTAS (as limited by ref 1, app 1). Gross weight was varied from 2490 to 2980 pounds, and density altitude was varied from minus 1600 to 10,530 feet. The results of the tests are presented in figures 63 through 76, appendix IV.

Table 5. Sideward and Rearward Flight Test Configurations.

Configuration <sup>1</sup>	Cross Weight (lb)	Density Altitude (ft)	Center of Gravity (in.)
Armed <sup>2</sup>	2,630	3,770	106.9 (fwd)
Armed <sup>2</sup>	2,680	-10	106.5 (fwd)
Armed <sup>3</sup>	2,490	10,530	108.7 (mid)
Armed <sup>3</sup>	2,980	150	107.5 (fwd)
Armed <sup>3</sup>	2,960	-1,600	106.2 (fwd) -2.4 (left lat)
Clean <sup>3</sup>	2,675	-970	106.4 (fwd)

<sup>1</sup>Doors on.

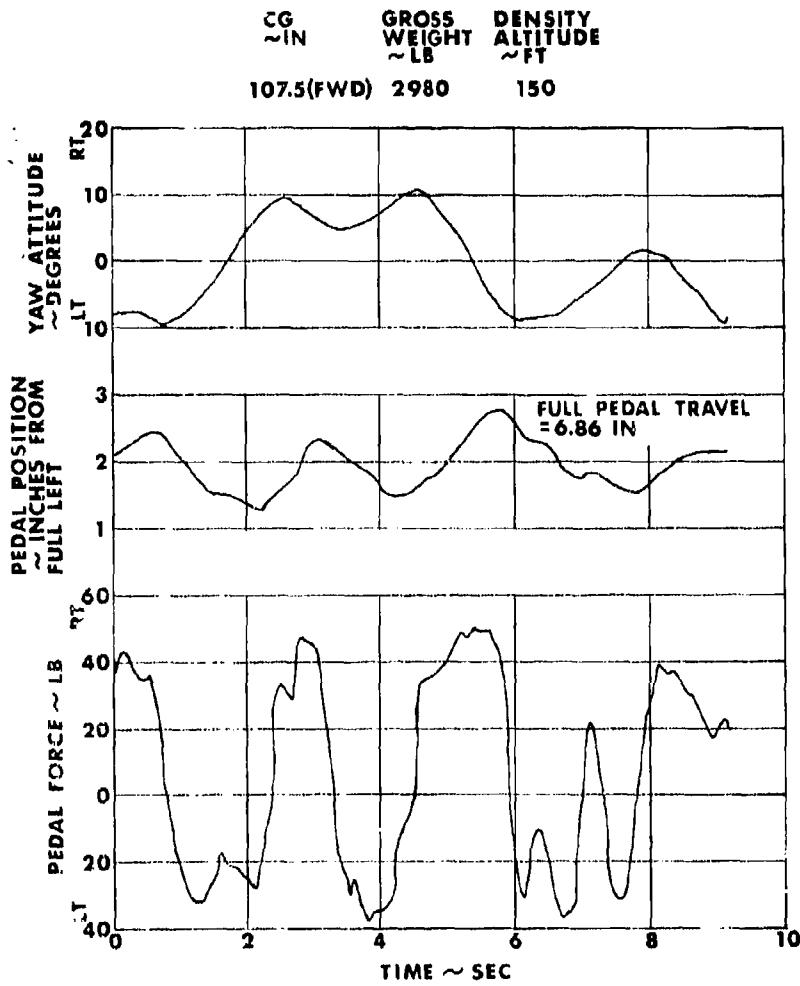
<sup>2</sup>Azimuth sweep (30-degree increments).

<sup>3</sup>Sideward, rearward and forward (90-degree increments).

52. The capability of hovering in crosswinds of various speeds and azimuths was adequate for most of the conditions tested. However, in left sideward flight at speeds from 15 to 25 KTAS and wind azimuths of 240 and 270 degrees, the helicopter was directionally unstable

necessitating large pedal force inputs to maintain a steady heading (HQRS 6). Time histories of left sideward flight at 15 KTAS are shown in figure G and also in figure 76, appendix IV. At airspeeds greater than 25 KTAS, the instability was not evident. All control margins were adequate for both right and left sideward flight up to 35 KTAS and complied with the requirements of the mil spec (as amended by ref 1, app I). The requirement of paragraph 3.3.5 of the mil spec (to execute a complete turn in each direction in a 35-knot wind) was not checked because of the unavailability of a 35-knot wind. However, the sideward and rearward flight data indicate that the aircraft does comply with this requirement. The directional instability in left sideward flight at 15 to 25 KTAS is a shortcoming, and correction is desirable.

**FIGURE G**  
**LEFT SIDEWARD FLIGHT AT 15 KTAS**



53. The position of the cyclic control was uncomfortably aft at the higher rearward flight speeds. At 30 KTAS with the doors on, the cyclic came in contact with the aft control stop several times while the pilot was correcting for minor longitudinal disturbances. The average margin of aft longitudinal control remaining reached the amended specification limit (ref 1, app I) of 5 percent at 30 KTAS, forward cg but never exceeded the limit (fig. 75, app IV). It is doubtful that the helicopter could hover in a forward cg doors-on configuration with a tail wind that exceeds 30 KTAS and still have the control margin required to overcome possible nose-down pitching. Hovering at such conditions is, therefore, not recommended. It is recommended that a caution note be placed in the operator's manual (ref 5) warning against hovering in a tail wind in excess of 30 knots.

54. There was no resultant change of any control margin caused by the removal of the gun. For all conditions tested, the only effect of moving the lateral cg to the full left location was that the cyclic control had to be moved 1.7 inches farther to the right than during the mid lateral cg location testing.

55. The effects of gross weight and density altitude changes were determined by varying the thrust coefficient ( $C_T$ ). At the greatest  $C_T$  tested (0.00343), an additional 0.7 inch of left pedal displacement was required for all flight conditions as compared with the lowest  $C_T$  of 0.00260. At 35 KTAS in right sideward flight at a  $C_T$  of 0.00343, the left pedal control margin reached 5 percent (fig. 72, app IV). At a greater  $C_T$  (increased grwt or higher  $H_D$ ), it would probably be impossible for the aircraft to comply with mil spec control margin requirements for right sideward flight at 35 KTAS. The right sideward flying qualities are satisfactory for normal operation.

56. The wind azimuths at which the control margins reached their lowest points are listed in table 6. The critical azimuths (headings of the helicopter relative to the wind which provided the minimum control margins) were identical for all test configurations listed in table 5. Typical results are shown for one configuration in figures 63 through 69, appendix IV.

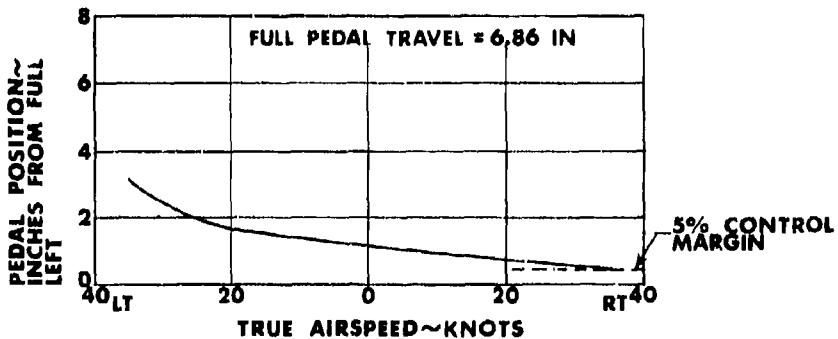
Table 6. Critical Azimuths.

Control	Flight Condition	Critical Azimuth (deg)	Control Margin Remaining (%)	True Airspeed (kt)
Left pedal	Right sideward	90	5	35
Aft cyclic	Rearward	180	5	30

57. Figures H and J show the pedal and longitudinal stick positions at the critical azimuth conditions where the control margins reached the amended specification limit of 5 percent (ref 1, app I). These margins were encountered only during the most critical conditions tested (cg, grwt,  $H_D$ , and aircraft configuration).

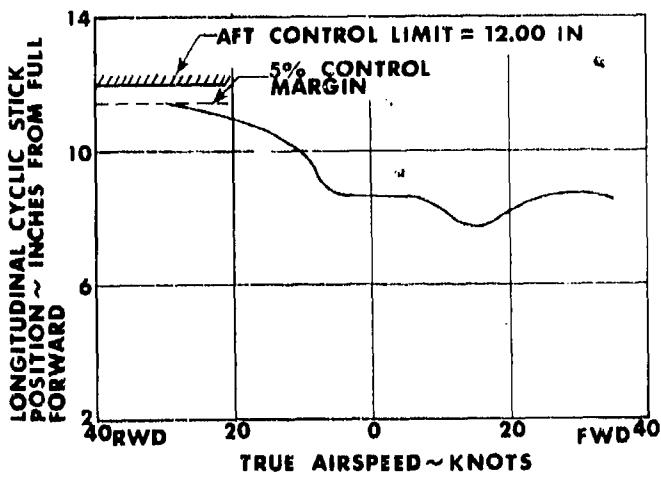
**FIGURE H  
SIDEWARD FLIGHT**

CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT
108.7(MID)	2490	10530



**FIGURE J  
REARWARD AND FORWARD FLIGHT**

CG ~IN	GROSS WEIGHT ~LB	DENSITY ALTITUDE ~FT
106.4(FWD)	2675	-970



58. A longitudinal stick reversal, which indicates a lack of static longitudinal trim stability, occurred from 20 to 30 KTAS in low-speed forward flight. The longitudinal stick position shifted aft 0.6 to 0.9 inch, depending on the configuration, as airspeed was increased. The longitudinal stick-position gradient changed sharply from nearly neutral at airspeeds between 5 KTAS rearward to highly negative (-0.43 in./kt) at airspeeds between 5 and 10 KTAS rearward. At rearward airspeeds exceeding 10 KTAS, the gradient was slightly negative (-0.037 in./kt). These discontinuities were not objectionable to the pilot and are not considered to be shortcomings.

#### AUTOROTATIONAL ENTRY

59. Simulated engine-failure tests (throttle chops) were conducted in level flight and during maximum-power climbs and maximum-power dives up to the never exceed airspeed ( $V_{NE}$ ) in the armed, doors-on configuration. The helicopter was trimmed at a given flight condition, and the throttle was abruptly closed to the flight idle position to simulate a sudden engine failure. The flight controls were held fixed as long as possible (up to a maximum of 2 seconds) to simulate the normal delay in pilot reaction time following an actual engine failure. The resultant maximum pitch, roll and yaw rates were recorded and plotted, as were the rotor decay rate and the time delay. The results of these tests, which include a time history plot, are presented in figures 77 through 80, appendix IV. The flight conditions tested are listed in table 7.

Table 7. Autorotational Entry Test Conditions.

Center of Gravity (in.)	Gross Weight (lb)	Density Altitude (ft)	Calibrated Airspeed Range (kt)
106.3 (fwd)	2720	3150	35 to 129
112.0 (aft)	2920	2250	35 to 129
112.4 (aft)	2920	9850	35 to 128

60. Generally, the reaction of the helicopter following a throttle chop was characterized by a slight nose-up pitch during the first second followed by a substantial nose-down pitching motion. The initial upward pitching motion was hardly noticeable in flight and did not exist at the highest airspeed (129 KCAS), forward cg condition. The secondary downward pitching was more pronounced, particularly at the greater airspeeds.

61. The reaction of the helicopter following the throttle chops was also characterized by a left yaw and left roll in all instances. The roll rates increased with increasing airspeed, while the yaw rates decreased as airspeed was increased. Yaw rates at the lower airspeeds (during climb and level-flight testing) were high and exceeded 20 deg/sec at the lower altitudes. Control effectiveness in recovery was adequate in all instances.

62. It was seldom possible to wait for a full 2-second delay between "rolling off" the throttle and lowering the collective control because of the rapid rate of rotor decay. In all instances, the collective was held fixed until it became necessary to lower it in order to arrest the rotor decay rate at or just above the minimum safe transient rotor speed of 304 rpm. The time required to fully reduce the collective averaged 0.5 second during which an additional 10 rpm were lost. This time is not included in the delay times as presented in this report. At a 2250-foot  $H_D$  (the lowest test altitude), the maximum acceptable time delay before application of control was 1.86 seconds (fig. 78, app IV). The minimum time delay was recorded at a 2250-foot  $H_D$  and was 1.16 seconds (fig. 78) during a maximum-power climb at 49 KCAS. At the 2250-foot  $H_D$  and at a 3150-foot  $H_D$ , the 2-second time delay requirement of paragraph 3.5.5 of the mil spec could not be met. At a 9850-foot  $H_D$ , the mil spec requirement (para 3.5.5) was met only for level-flight speeds of less than 80 KCAS.

63. At all altitudes, the low-airspeed (approximately 49 KCAS) maximum-power climbs resulted in the highest rate of rotor decay after the throttle chop and required rapid corrective action by the pilot. Maximum delay time at this flight condition was 1.63 seconds at a 9850-foot  $H_D$  (fig. 79, app IV). After the minimum rpm was reached, the rotor speed recovery rate (rotor speed build-up to obtain a normal rate after the collective is lowered) was slower from climb throttle chops than from the level-flight and/or dive conditions. In level flight, the rotor speed decay rate increased as airspeed increased, but the pilot response time was not as critical as it was during climb.

64. Warning of engine failures was adequate for all conditions tested. There was a distinct decrease in sound level as the throttle was closed which was accompanied by an instantaneous yaw to the left. The slight instantaneous pitching motion did not provide an adequate cue. There were no side forces or normal acceleration forces (positive or negative) of sufficient magnitude to be objectionable. Collective coupling was not observed. The average delay time for all conditions tested was acceptable, but rapid pilot compensation was required in climb. Rotor speed response to variations in collective setting and airspeed was sensitive and required constant monitoring. Rotor speed fluctuations were not large, however, and an inexperienced pilot in an actual emergency would probably have no difficulty keeping

the rotor speed within limits. Although considerably more aft cyclic was required in autorotation than in level flight (para 21), control margins were ample. The overall autorotational entry characteristics of the OH-58A are satisfactory (HQRS 3).

#### HOVERING STABILITY

65. The stability of the OH-58A helicopter in a hover was satisfactory for the pitch and roll axes but was unsatisfactory in yaw. Excessive pilot effort was required to maintain precise directional control and to make heading changes. Control inputs which were required to compensate for heading disturbances often resulted in moderate pilot-induced oscillations (PIO). The severity of the PIO was directly related to the heading accuracy required. The lack of yaw-rate damping (para 43) required the pilot to make control inputs to compensate for any yaw disturbance. The pilot's task in maintaining a heading was complicated by a high pedal breakout force of 10 to 15 pounds (measured in flight) which is significantly out of proportion to the breakout force in the lateral and longitudinal control systems (0.5 lb). In addition to the large breakout force, the pilot encountered a transient pedal force gradient which was present only while the pedals were moving. This transient force gradient reached values on the order of 100 lb/in. These factors combine to complicate the pilot's task of holding the helicopter steady while in a hover and during landing. The net pedal forces were measured by strain gauges and recorded on the oscillograph (figs. 76 and 90, app IV).

66. Landing in confined areas, such as revetments and unimproved landing zones, requires precise directional control. Excessive pilot effort required to accomplish this basic task detracts from the helicopter ability to perform operational missions. The shortcoming in directional control during hover (HQRS 5) also detracts from the helicopter accuracy during firing of the armament subsystem and could adversely affect mission accomplishment. Correction is recommended on a priority basis.

#### ARMAMENT FIRING

67. The XM27E1 armament subsystem was fired at representative conditions of flight to determine its effect on the stability and control of the helicopter. The results are presented in figures 81 through 84, appendix IV. Reaction to firing the weapon resulted in attitude changes in pitch, roll and yaw. These attitude excursions and the imprecise helicopter response to pilot efforts in correcting the alignment of the weapon with the target degraded the capability of the helicopter to maintain impact in the target area.

68. Firing was conducted in two phases. During Phase I, the flight controls were held fixed while firing 3-second bursts. These data are presented in figure 81, appendix IV. In Phase II, corrections were applied in an attempt to maintain accurate target impact. Time histories of this firing are presented in figures 82 through 84. Qualitative evaluations of pilot effort, noise and vibration level were made. Time histories of control positions, helicopter attitudes and rates were recorded on an oscilloscope. The duration of the firing burst was restricted to 3 seconds by a limiter built into the firing circuit.

69. Firing runs were made from a low level (approximately 50 feet) and during dives from 1500 feet above ground level (AGL). Each firing run was followed by a steep climbing turn to the right. The right turn was considered to be the most critical condition for control-margin evaluation. Firing conditions are presented in table 8.

70. Firing from a hover caused the most problems for the pilot. Precise aiming was extremely difficult because of a tendency to overshoot the desired point when attempting to make small heading corrections. This problem was compounded during firing by the large reaction in the pitch, roll and yaw axes resulting from weapon firing. When corrections were not applied to hold the aircraft attitude, the heading changed 47 degrees (to the left), and the nose pitched down 17 degrees at the end of a 3-second burst at the maximum rate of fire (4000 rds/min) with the weapon horizontally aligned (fig. 81, app IV). The helicopter had adequate control power available to maintain control and to correct the target alignment, but the high yaw rate (reaching a maximum of 24 deg/sec to the left), resulting from the firing, and the difficulty encountered in maintaining precise directional control prevented hitting the target consistently (fig. 84). Correction caused the hits to "walk" across the target, and the hits could not be held within the target area consistently.

71. Targets were engaged in forward flight at representative air-speeds at low altitudes and also during dives from 1500 feet AGL. Each target attack run was followed by a steep climbing turn up to bank angles of 60 degrees. Despite the roll that resulted from firing, adequate control margin was available for executing evasive maneuvers.

Table 8. Firing Test Conditions.

## Phase I (Controls Fixed)

Calibrated Airspeed (kt)	Flight Condition	Height Above Ground (ft)	Weapon Position	Rate of Fire (rd/min)
0	Hover TGE	10	Full up, horizontal, full down	2000
0	Hover TGE	10	Horizontal	4000
0	Hover OGE	50	Full up, horizontal, full down	2000
0	Hover OGE	50	Horizontal	4000
49	Level flight	1500	Horizontal	2000 and 4000
80	Level flight	1500	Full up, horizontal, full down	2000 and 4000
80	5, 10 and 25 degrees (left and right sideslip)	1500	Horizontal	2000
80	Accelerated flight at 1.4 g's	1500	Full up, horizontal, full down	2000
80	Accelerated flight at 1.4 g's	1500	Horizontal	4000
80	Accelerated flight at 1.8 g's	1500	Full up, horizontal, full down	2000
80	Accelerated flight at 1.8 g's	1500	Horizontal	4000
105	Level flight	1500	Horizontal	2000 and 4000
117	Dive	1500 to 500	Horizontal	2000 and 4000
122	Dive	1500 to 500	Horizontal	2000 and 4000

Phase II (Controls Applied)

Calibrated Airspeed (kt)	Flight Condition	Height Above Ground (ft)	Weapon Position	Rate of Fire (rd/min)
0	Hover IGE	10	Horizontal	2000 and 4000
0	Hover IGE	10	Full down	2000
0	Hover OGE	50	Full down, horizontal	2000 and 4000
33	Low level	50	Horizontal	2000 and 4000
49	Low level	50	Horizontal	2000 and 4000
85	Low level	50	Horizontal	2000 and 4000
101	Dive	1500 to 500	Horizontal	2000 and 4000
125	Dive	1500 to 500	Horizontal	2000 and 4000
125	Dive	1500 to 500	Full down	4000
10 to 30	Takeoff	5	Horizontal	2000 and 4000
10 to 30	Takeoff	5	Full up	2000
30 to 10	Landing	5	Horizontal	2000 and 4000
30 to 10	Landing	5	Full down	2000
15	Left sideward	10	Horizontal	2000 and 4000
15	Right sideward	10	Horizontal	2000 and 4000
15	Rearward	10	Horizontal	2000 and 4000

72. Difficulty was encountered in establishing precise aim before firing because of excessive vertical vibration of the sight reticle (HQRS 5) and the difficulty in making small directional corrections. Use of the lateral cyclic control alone to make directional corrections was not suitable because of the excessive lapsed time before the turn could be completed (approximately 2 seconds). Attempts to coordinate pedal and cyclic control required excessive pilot effort in order to maintain balanced flight because of the disharmony.

of the control forces (disproportionate pedal forces). The use of the pedals alone resulted in excessive sideslip. The difficulty in holding precise headings makes the consistency of achieving first round hits on a target very poor. Sight reticle vibration is a shortcoming which would adversely affect mission accomplishment, and correction is recommended.

73. Attitude excursions which resulted from firing during forward flight were not large but did cause significant dispersion of hits (figs. 82 and 83, app IV). Control power was adequate for control of the helicopter and to correct for attitude disturbances, but it was impossible to maintain impact on a point target.

74. Noise and vibration levels were evaluated qualitatively based on their effects on the pilot's ability to perform the mission. During firing, the increase in the vibration level was moderate and did not seriously affect the flying qualities of the helicopter (HQRS 3). The increase in noise level was high during firing, particularly in the doors-off configuration. Excessive noise in the intercom made conversation difficult to understand over the intercom and would make radio transmission difficult during a combat situation. Correction of this shortcoming is recommended.

75. Changes in helicopter attitude which resulted from firing were related to airspeed, rate of fire and weapons position (relative to the horizontal axis) as shown in figure 81, appendix IV. The magnitude of the changes decreased with increasing airspeed and was less at the reduced rates of fire. Firing of the weapon in the horizontal or elevated positions resulted in a left yaw, left roll, and a nose-down pitch attitude. Firing of the weapon in the depressed position resulted in a left yaw, right roll, and nose-down pitch attitude.

76. Adequate control power was available to correct for the attitude changes observed in all flight conditions tested, and no change in the stability characteristics was noted; however, excessive pilot effort was required to correct the point of impact back to the target. In addition, a high level of pilot effort was required to make minor directional aiming corrections prior to initiating firing (HQRS 5). Correction of the difficulty in making precise heading corrections is recommended to improve service suitability.

#### FLIGHT CONTROL SYSTEM

77. The flight control system was evaluated to determine the control breakout forces (total force, including friction, required to initiate control movement), force gradients and maximum flight

control forces, and also to determine compliance with the requirements of the mil spec. The system was tested with and without the hydraulic boost system in operation. The breakout forces and force gradients were measured on the ground with the helicopter rotor stationary using hydraulic pressure applied from an external source. Although the mil spec breakout force requirements are specified in terms of in-flight conditions, common test procedure has been to consider ground measurements valid. The boost-OFF forces were also measured during normal flight maneuvers (see time histories of symmetrical pull-up, left sideward flight and landing, figs. 62, 76 and 90, app IV). The results are presented in figures 85 through 90.

78. The boost-ON longitudinal breakout forces (fig. 86, app IV) were measured with the force trim switch both ON and OFF and with the cyclic friction OFF. The force-trim-ON, friction-OFF breakout force was 1.7 pounds (pull force) which slightly exceeded the 1.5-pound limit imposed by paragraph 3.2.7 of the mil spec. Longitudinal breakout forces in flight were not objectionable and allowed small, smooth, precise control displacements from trim. All longitudinal force gradients (force trim ON) were positive. With cyclic friction applied during a normal flight adjustment, the force-trim-ON stick force gradient was smooth, and there were no discontinuities recorded.

79. The lateral breakout forces (boost ON) were measured (fig. 87, app IV) with the force trim both ON and OFF using a normal flight application of cyclic friction (force trim ON). The maximum lateral breakout force (force trim ON, friction OFF) was 0.8 pound and complied with the requirements of paragraph 3.3.13 of the mil spec. The force gradient was positive at all times, and the requirements of paragraph 3.3.11 of the mil spec were met. No undesirable discontinuities in the lateral force gradient were observed during flight. There was no binding in the system with cyclic friction applied.

80. The maximum pedal breakout force (fig. 85, app IV) measured on the ground was approximately 9 pounds (right) and exceeded the 7-pound limitation of paragraph 3.3.13 of the mil spec. The directional control force gradient was nonlinear over the entire trim range and was negative to the right of the trim position. These results do not meet the requirements of paragraph 3.3.11 of the mil spec.

81. Pedal breakout forces varied with the flight condition. They were approximately the same in level flight as those measured on the ground, whereas the forces were considerably greater during hovering flight as can be seen in a time history of a typical landing approach (fig. 90, app IV). The pedal forces reached a maximum transient value of 49 pounds during left sideward flight at 15 KTAS (fig. 76). This force is considered to be excessive and, therefore, did not meet the

qualitative requirements of deviation 19 of the detail specification. The 15-pound directional control force limitation of paragraph 3.3.12 of the mil spec was not met.

82. Pedal free play (pedal travel necessary to cause a deflection of the tail-rotor control surface) was measured on the ground and was found to be  $\pm 0.1$  inch which meets the requirements of paragraph 3.5.10 of the mil spec. However, the total effect of this free play in conjunction with the excessive pedal breakout forces was considered to be a contributing factor to the directional control in a hover (discussed in para 65).

83. The maximum collective breakout force (fig. 88, app IV) (boost ON, friction OFF) measured at a typical collective setting of 5 inches was approximately 4.3 pounds (pull). Although this exceeded the requirements of paragraph 3.4.2 of the mil spec by 1.3 pounds, the boost-ON collective breakout force was not considered to be objectionable in flight.

84. Some control coupling (boost ON) existed between the collective and the longitudinal control (fig. 89, app IV). When the collective was moved upward, a 0.5-pound forward force was transmitted to the cyclic control. Although this force coupling did not meet the requirements of paragraph 3.4.3 of the mil spec, it was not objectionable to the pilot because the use of a small amount of cyclic friction eliminated the effect.

85. The longitudinal force gradient was approximately 1.42 lb/in. as compared with a lateral gradient of 0.88 lb/in. (boost ON, friction OFF). The gradients did not change when cyclic friction was applied. Control harmony among the lateral, longitudinal and collective controls was satisfactory except in those instances in which high pedal forces were encountered.

86. The longitudinal control free play (boost OFF) was measured and found to be  $\pm 0.53$  inch, and the lateral free play (boost OFF) was found to be  $\pm 0.91$  inch. These free-play measurements exceeded the limitations of paragraph 3.5.10 of the mil spec by 0.33 inch and 0.71 inch, respectively.

87. Excessive coupling between the cyclic and collective controls existed with the boost OFF (fig. 89, app IV). Raising the collective control caused a maximum pull force of 6 pounds to be transmitted to the cyclic control which exceeded the 1-pound maximum limitation of paragraph 3.4.3 of the mil spec. Lowering of the collective caused a push force of 4.2 pounds to be transmitted to the cyclic control.

88. All of the maximum breakout forces (boost OFF) were found to be excessive (fig. 85, app IV). The maximum longitudinal force was measured at 14.5 pounds (pull); the maximum lateral force was 13 pounds (left from neutral); and the maximum collective force was 38 pounds (up from the 5-inch collective setting). These forces are shown in time history plots of the data which were measured in flight (figs. 62 and 90), and each exceeded the limits of paragraphs 3.2.7, 3.3.13 and 3.4.2 of the mil spec. The boost-OFF lateral, longitudinal and collective breakout forces were considered acceptable for emergency use only (HQRS 6).

#### VIBRATION

89. Vibrations in the helicopter were measured at the various flight conditions listed in table 9. Accelerometers were installed at the pilot seat and the passenger-seat to measure lateral and vertical vibrations. Vibration frequencies were analyzed at multiples of one-, two-, four-, six- and eight-per-rotor-revolution. Single-amplitude vibration and acceleration levels were plotted as a function of airspeed and are presented in figures 91 through 95, appendix IV. Maximum recorded vibration levels are listed in table 9.

90. All lateral vibration levels were insignificant (0.11 g or less) and complied with the requirements of the mil-spec. The predominant vertical vibration was the two-per-revolution which increased significantly with increasing airspeed at airspeeds greater than 50 KCAS. The vibration level at this frequency, as well as the six-per-revolution vibration, exceeded the limit of paragraph 3.7.1(b) of the mil spec for several conditions listed in table 9. The two-per-revolution vibration became severe as  $V_{NE}$  was approached. Although this provided an ample cue to warn the pilot of impending blade stall, the vibration level near  $V_{NE}$  was uncomfortable and degraded the target tracking task (HQRS 4). Correction of this shortcoming is recommended for improved service use. Increasing the maximum level-flight speed capability of the OH-58A to speeds more closely approaching  $V_{NE}$  would probably increase the vibrations to a level which would be unacceptable for sustained level-flight operations. At airspeeds of 100 KCAS and less, the vibration levels were satisfactory (HQRS 3). Varying the flight conditions or removing the armament subsystem did not significantly affect the vibration levels. Vibration levels in climb and autorotation did not vary significantly from those recorded during level flight.

Table 9. Flight Conditions and Maximum Vibration Levels.

Gross Weight (lb)	Center of Gravity (in.)	Density (ft)	Calibrated Airspeed (kt)	Configuration: Clean	Vertical Vibration			
					Pilot Seat	Passenger Seat	Military Specification Limit <sup>2</sup> (g)	2/rev (g)
2,750	107.6 (Fwd)	5,780	<sup>3</sup> 119.0	Clean	0.22	0.14	0.29	0.18
2,710	107.0 (Fwd)	5,000	<sup>3</sup> 125.0	Armed	0.25	0.12	0.26	0.15
2,985	106.3 (Fwd)	4,580	<sup>3</sup> 115.0	Armed	0.19	0.16	0.20	0.22
2,995	112.2 (aft)	4,830	<sup>3</sup> 115.0	Armed	0.26	0.13	0.17	0.23
2,730	107.1 (Fwd)	10,950	<sup>4</sup> 105.0	Armed	0.21	0.12	0.21	0.16
2,750	107.6 (Fwd)	5,780	<sup>5</sup> 85.0	Clean	0.09	0.04	0.10	0.19
2,710	107.0 (Fwd)	5,000	<sup>5</sup> 85.0	Armed	0.10	0.06	0.09	0.14

<sup>1</sup>Doors on.<sup>2</sup>MIL-H-8501A.<sup>3</sup>Dive.<sup>4</sup>Level flight.<sup>5</sup>Autorotation.

### SLOPE LANDING

91. An evaluation was conducted to investigate the slope-landing characteristics of the OH-58A. It was impossible to land safely on a 10-degree slope (measured with an inclinometer) with the left skid uphill. The helicopter could be landed safely with the nose, tail and right skid uphill on a 10-degree slope, although the latter orientation required the application of full right cyclic control. It is doubtful that the maneuver would be acceptable to the average pilot who would probably abort the landing when full lateral travel of the cyclic had been reached. When the left skid was uphill, the maximum slope capability was approximately 9 degrees.

92. The landing surface at the test site consisted of a dry, well-graded, firmly compacted, clay/sand desert soil with an approximate California bearing ratio of 10 and a 25-degree angle of repose. Surface winds during the test were steady at 10 to 15 knots and provided a head-wind condition when the left side of the helicopter was "upslope." The method of test was to perform landings on progressively steeper slopes using cyclic control as required to hold the aircraft against the slope while the collective pitch was lowered smoothly and gradually. When the skids were firmly set on the ground with no tendency to slip, the collective control was placed in the full down position, and the cyclic control was centered. The reverse sequence was used to lift off and return to a hover. In the left-skid-uphill test, the left skid tended to slide down the 10-degree slope despite the use of full left lateral cyclic and the lowering of the collective control to the point where the downhill (right) skid was still 4 to 5 inches from the ground.

93. The requirements of paragraph 3.8.1 of the detail specification were not met in that the helicopter could not land safely on a 10-degree slope from all directions. It should be noted, however, that several possible variables could have affected the slope-landing capabilities: soil type and condition, wind, and pilot technique. Since these variables are not defined in the detail specification, the results of the test are not conclusive evidence that the requirements cannot be met.

### SKI INSTALLATION

94. A brief qualitative evaluation was conducted to determine the OH-58A flying qualities with Airglas Corporation skis (model no. L2700-206) installed. The ski installation is shown in photo 1. The following tests were conducted at the maximum airspeed for level flight ( $V_H$ ) in the armed configuration: static and dynamic longitudinal, lateral and directional stability, maneuvering stability

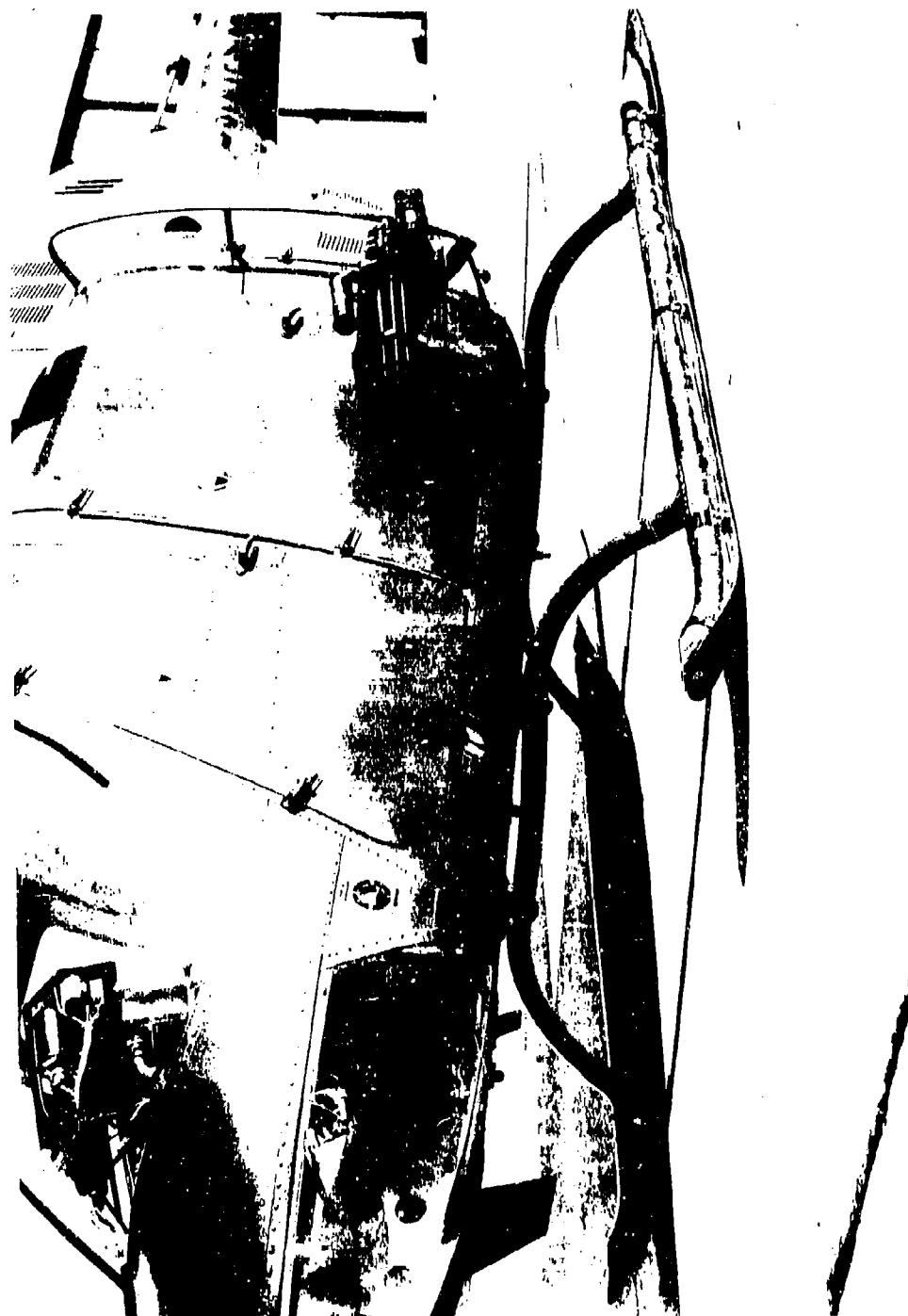


Photo 1. Ski Installation.

and autorotational entry. A slight two-per-revolution vibration was observed in the skis, particularly the left one. This vibration had no discernible effect on the flying qualities or safety of the helicopter, and the evaluation confirmed the ability of the OH-58A to fly safely with skis installed.

#### HUMAN FACTORS

95. There were no hand holds installed on the helicopter to assist the crew members in entering and leaving the cockpit. Equipment improvement recommendation (EIR) number 619269 was submitted on 5 March 1970 regarding this problem. Ingress and egress were difficult, particularly for a tall individual, because of the size and shape of the door. Correction of this shortcoming is recommended to facilitate crew ingress and egress.

96. Ingress and egress were further impeded by the lack of hooks inside the cockpit on which to hang the pilot's and copilot's helmets. The pilot and copilot must place their helmets either directly in the way on the seat or floor, or depend upon another individual outside the helicopter for assistance. Correction of this shortcoming is recommended.

97. All four doors had jettison releases. The handles were marked adequately and could be operated easily. During a firing run, the left front door came off at the emergency release points, and the cause was not determined. EIR number 143940 and EIR number 54574 were submitted regarding this problem on 13 January and 17 February 1970, respectively. The emergency door release was satisfactory for emergency egress.

98. The mechanism designed to hold the doors in place when opened during ground operation was unsatisfactory. It was usually impossible to keep the doors open without physically holding them. Correction of this shortcoming is recommended.

99. The pilot and copilot seats were comfortable when parachutes were not worn. No seat adjustment was provided; therefore, when parachutes were worn, it was necessary to remove the back of the seat. In these cases, the pilot and copilot were still seated farther forward than the normal position. A tall pilot wearing a parachute had difficulty obtaining full right cyclic deflection because the door post restricted the movement of his right knee to the extent that his right leg was in the path of the cyclic control. Since parachutes are seldom worn during normal operation, the crew seats are satisfactory for mission accomplishment.

100. The antitorque pedals are adjustable by turning a knob located on the floor just aft of the pedals. The pedal adjustment knob was easy to turn by the pilot using his foot and permitted a satisfactory range of pedal travel for most pilots. The adjustment was adequate for mission accomplishment which is an important consideration because the seat is not adjustable.

101. The force trim button is located on the top right-hand corner of the cyclic stick grip (looking forward). Activation of the button required placement of the pilot's right thumb in an awkward position and resulted in a momentary reduction of control authority. It is recommended that this shortcoming be corrected by relocating the force trim button in accordance with the position of the standardized cyclic grip, MS87017.

102. The cyclic and collective friction adjustments were satisfactory. The throttle friction was excessive and was not adjustable. When recovering from practice autorotations, it was possible to inadvertently fail to apply full power because of the excessive friction. A throttle friction adjustment device is recommended to correct this shortcoming.

103. There is no map storage compartment in the front (pilot) cockpit. This is unsatisfactory in a warm weather environment where the helicopter is flown with the doors removed. Installation of a cockpit map storage compartment is recommended to correct this shortcoming.

104. The ventilation system, consisting of two snap vents in each of the four doors, was inadequate for warm weather operation with the doors installed. Based on crew observation, there were no engine fumes present in the cockpit during ground or air operations; however, the pilot would probably become fatigued because of the heat. Correction of this shortcoming is recommended.

#### MAINTENANCE CHARACTERISTICS

##### Favorable

105. In general, the maintainability of the helicopter was excellent throughout the test. "Down time," because of maintenance problems, was almost nonexistent. A total of 183 flight hours was accumulated on the test helicopter without the occurrence of a serious maintenance problem.

Unfavorable

106. Several unfavorable characteristics were noted; these are as follows:

- a. The ground handling wheels were inadequate. EIR Number 619268 was submitted on 5 March 1970 regarding this problem. A full fuel load would cause the wheels to spread out, and the tires would become almost flat despite a 10-psi overpressure. Correction of this shortcoming is recommended.
- b. Ground handling was impeded by the lack of handling points on the helicopter both on the tail section and on the fuselage intermediate or forward section. Correction of this shortcoming is recommended.
- c. During the high-altitude testing (15,000-foot  $H_D$ ), oil leaked through the outboard ends of the main rotor blade grips. Leakage was not excessive, but correction is recommended in order to preclude future problems during high-altitude operation. EIR Number 457831 was submitted on 15 October 1969 regarding this problem.
- d. The fuel quantity gauge indicated FULL when the VHF transmitter was keyed or when the force trim button was actuated. Correction of this shortcoming is recommended.
- e. The rubber bumper located next to the static stop on the tail-rotor assembly was replaced four times during the evaluation. This was a result of the static lateral-directional testing in which large sideslip angles were encountered and is not considered to be a shortcoming for the normal mission.

## **CONCLUSIONS**

### GENERAL

107. The following conclusions were reached as a result of the stability and control testing on the OH-58A helicopter:

- a. The overall flying qualities of the helicopter are satisfactory.
- b. There are 19 items of specification noncompliance. These are listed in appendix V.

### DEFICIENCIES AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

108. There were no deficiencies noted during the testing.

109. Correction of the following shortcomings is desirable for improved operation and mission capabilities:

- a. Dutch-roll tendency when the helicopter is in a slight left sideslip (para 30).
- b. Lack of yaw rate damping when the helicopter is in a hover (para 43).
- c. High boost-OFF collective and cyclic control forces (paras 50 and 88).
- d. Directional instability in left sideward flight at airspeeds between 15 and 25 KIAS (para 52).
- e. Difficulty in maintaining precise directional control in hover (para 66).
- f. Excessive vertical vibration of the sight reticle in the armament subsystem (para 72).
- g. Excessive noise in the intercom during firing (para 74).
- h. Difficulty in making precise directional control corrections in forward flight while aiming and firing the weapon system (para 76).
- i. Severe two-per-revolution vertical vibrations at speeds near VNE (para 90).

- j. Lack of hand holds on door frames (para 95).
- k. Lack of hooks in the cockpit on which to hang helmets (para 96).
  - l. Difficulty in keeping doors open during ground operations (para 98).
  - m. Awkward location of force trim button (para 101).
  - n. Excessive friction of the throttle and no adjustment (para 102).
  - o. Lack of a cockpit map storage compartment (para 103).
  - p. Inadequate cockpit ventilation (para 104).
  - q. Inadequate ground handling wheels (para 106a).
  - r. Lack of ground handling points (para 106b).
  - s. Oil leakage through main rotor blade grips at high altitude (para 106c).
  - t. Fuel quantity gauge indicated FULL when the UHF transmitter was keyed or the force trim button was actuated (para 106d).

## **RECOMMENDATIONS**

110. The shortcomings, correction of which is desirable for the improvement of the helicopter mission capabilities, should be corrected as soon as possible.

111. The directional instability in hover should be improved on a priority basis.

112. A caution note should be placed in the operator's manual warning against hovering in tail winds in excess of 30 knots.

## APPENDIX I. REFERENCES

1. Detail Specification, Bell Helicopter Company, 206-947-031, *Light Observation Helicopter*, Revision No. R-3, 15 January 1968.
2. Letter, AMCPM-LH-T, HQ, USAAVSCOM, subject: Test Directive, OH-58A Light Observation Helicopter Flight Test ATA Project No. 68-30, 7 August 1968.
3. Final Report, USAASTA; Project No. 68-30, *Airworthiness and Flight Qualification Test, Production OH-58A Helicopter, Unarmed and Armed with XM27E1 Weapon System, Performance*, June 1970.
4. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961, with Amendment 1, 3 April 1962.
5. Operator's Manual, TM 55-1520-228-10, *Army Model OH-58A Helicopter*, 30 June 1969, changed 1 November 1969.
6. Test Plan, USAASTA, Project No. 68-30, *Engineering Flight Test of the Production OH-58A Helicopter, Unarmed and Armed with XM27E1 Weapon Subsystem*, May 1969.

## **APPENDIX II. TEST INSTRUMENTATION**

Sensitive instruments were installed in the test helicopter and were maintained by the Instrumentation Branch of USAASTA. The following parameters were recorded:

### Pilot and Engineer Panels

Airspeed (boom system)  
Altitude (boom system)  
Angle of sideslip  
Rotor speed  
CG normal acceleration  
Free air temperature  
Longitudinal cyclic stick position  
Lateral cyclic stick position  
Collective stick position  
Rudder pedal position  
Total fuel used  
Oscillograph coordination counter

### Oscillograph

Longitudinal cyclic stick force  
Lateral cyclic stick force  
Collective stick force  
Rudder pedal force  
Longitudinal cyclic stick position  
Lateral cyclic stick position  
Collective stick position  
Rudder pedal position  
Pitch attitude  
Roll attitude  
Yaw attitude  
Pitch rate  
Roll rate  
Yaw rate  
Pitch angular acceleration  
Roll angular acceleration  
Yaw angular acceleration  
CG normal acceleration  
Angle of attack  
Angle of sideslip  
Linear rotor speed  
Rotor blip  
Torque pressure

Throttle position  
Engineer event  
Pilot event  
Pilot vertical vibration  
(FS = 73 in., WL = 29 in., butt line (BL) = 15 in.)  
Pilot lateral vibration  
(FS = 73 in., WL = 29 in., BL = 15 in.)  
Passenger vertical vibration  
(FS = 108 in., WL = 32 in., BL = 0 in.)  
Passenger lateral vibration  
(FS = 108 in., WL = 32 in., BL = 0 in.)  
Gun position  
Rate of fire

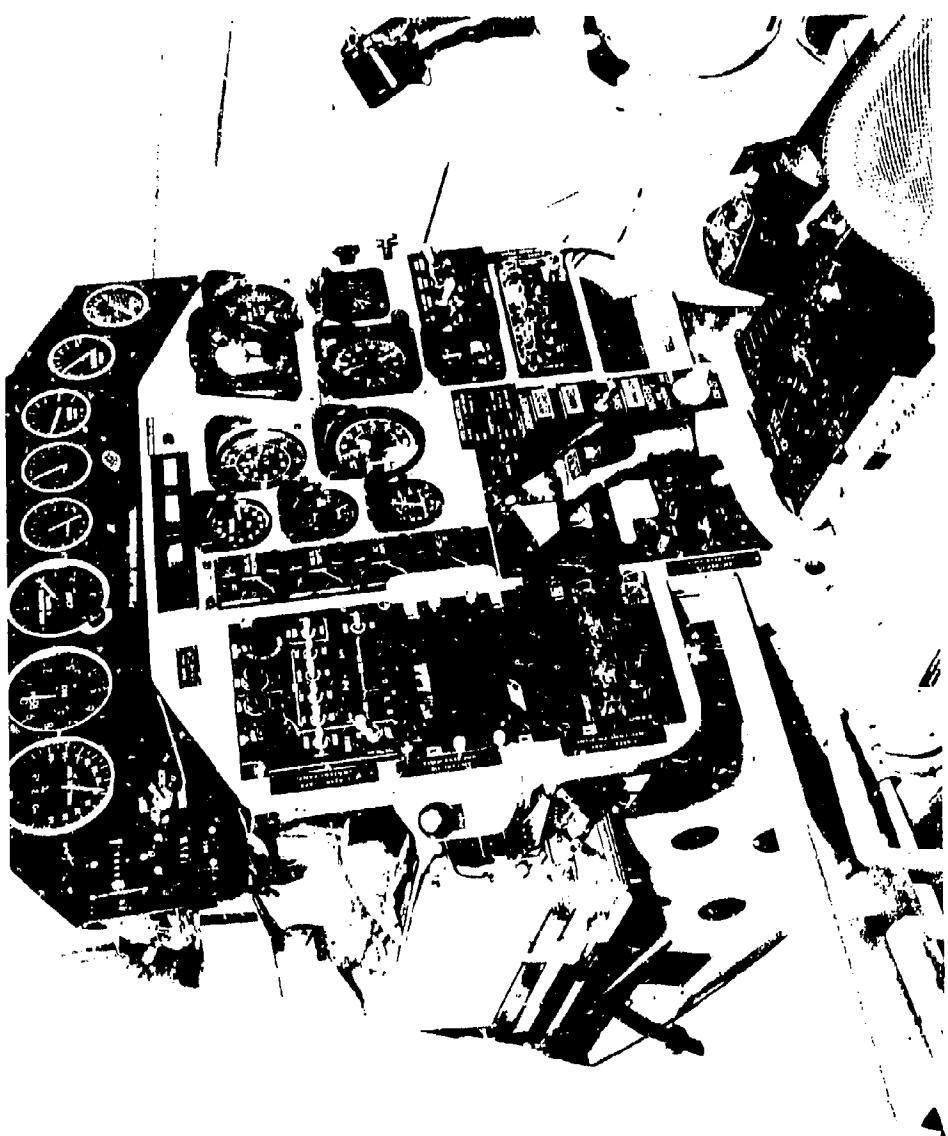


Photo A. Instrument Panel.

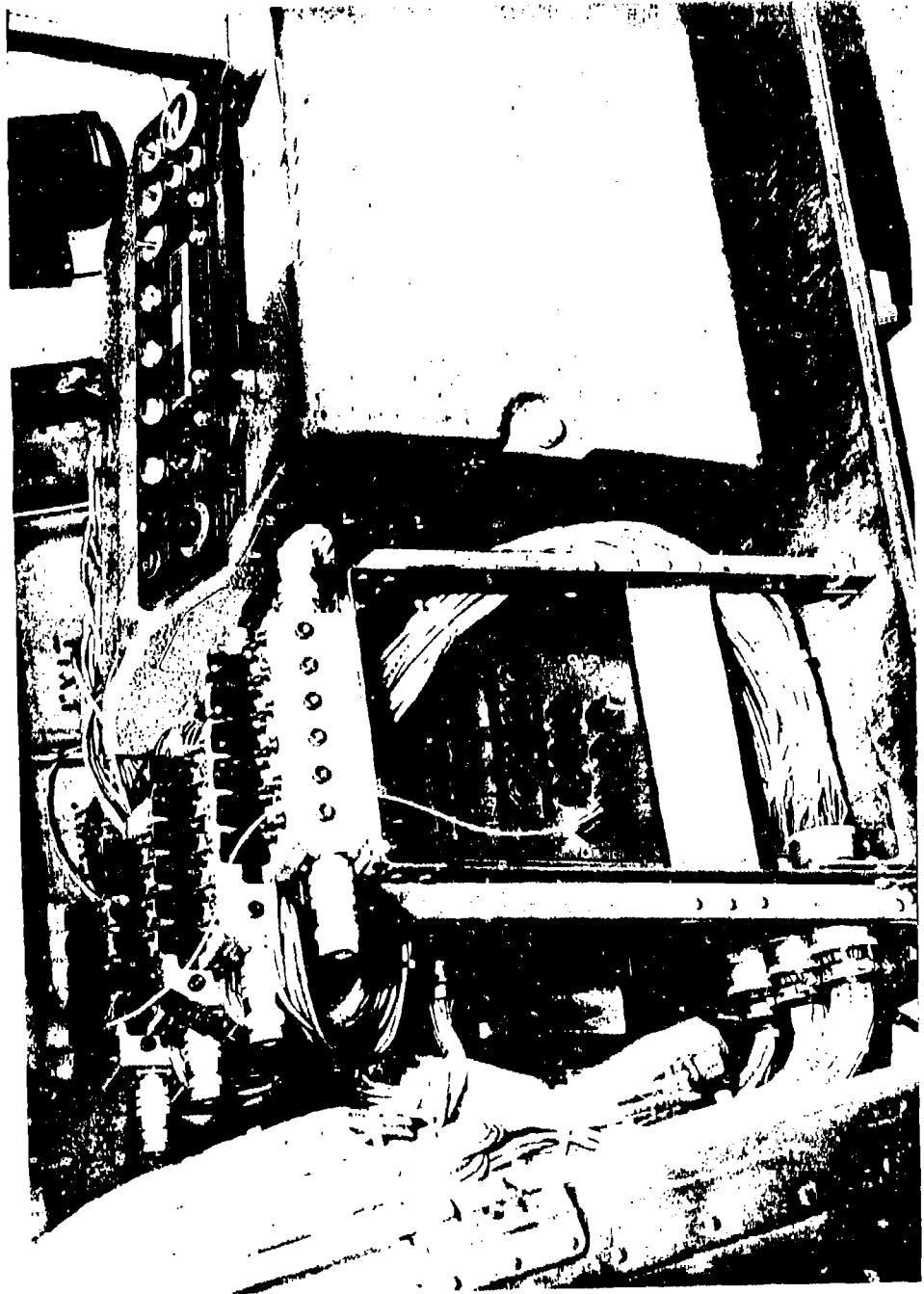


Photo B. Oscilloscope Installation.

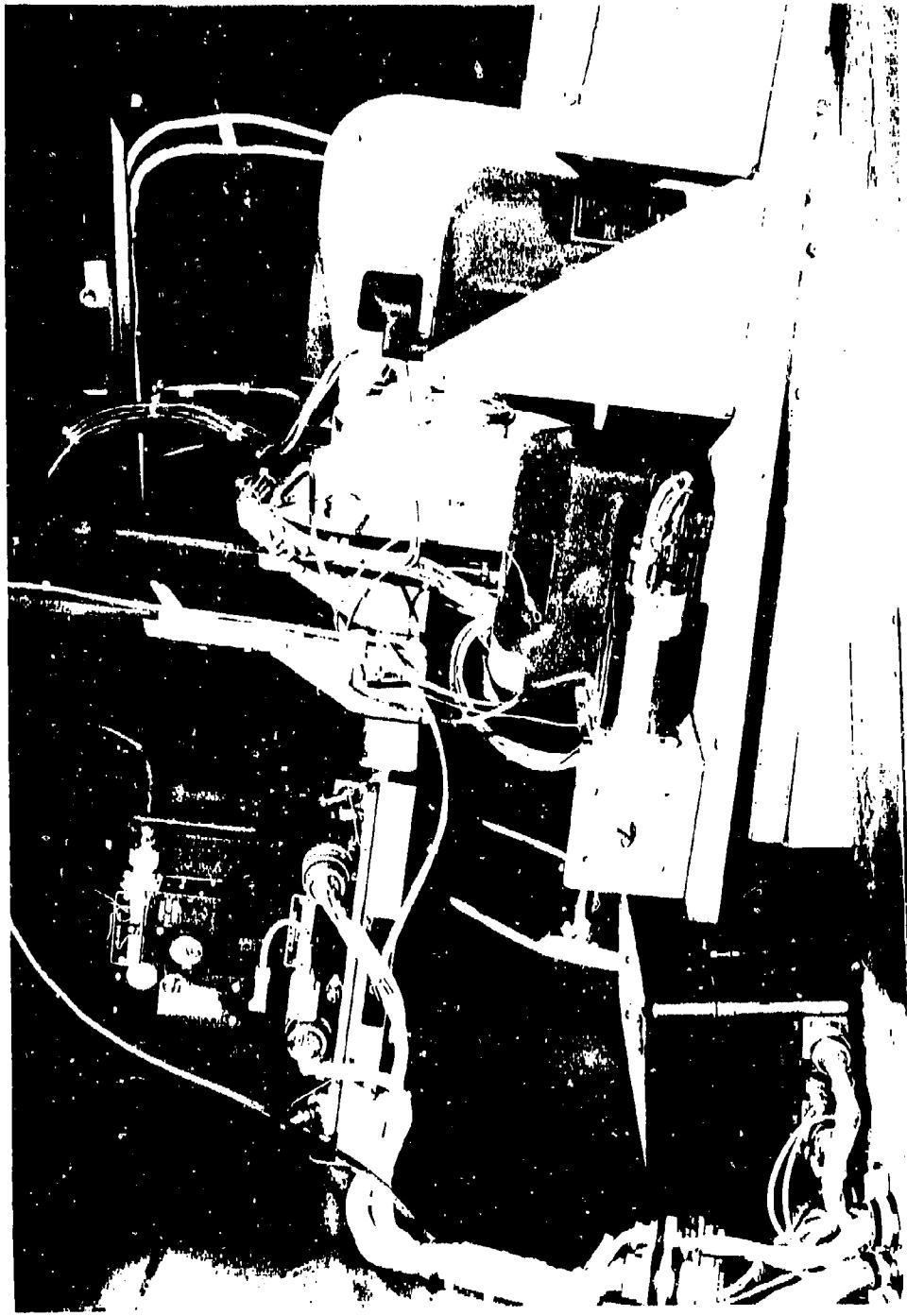
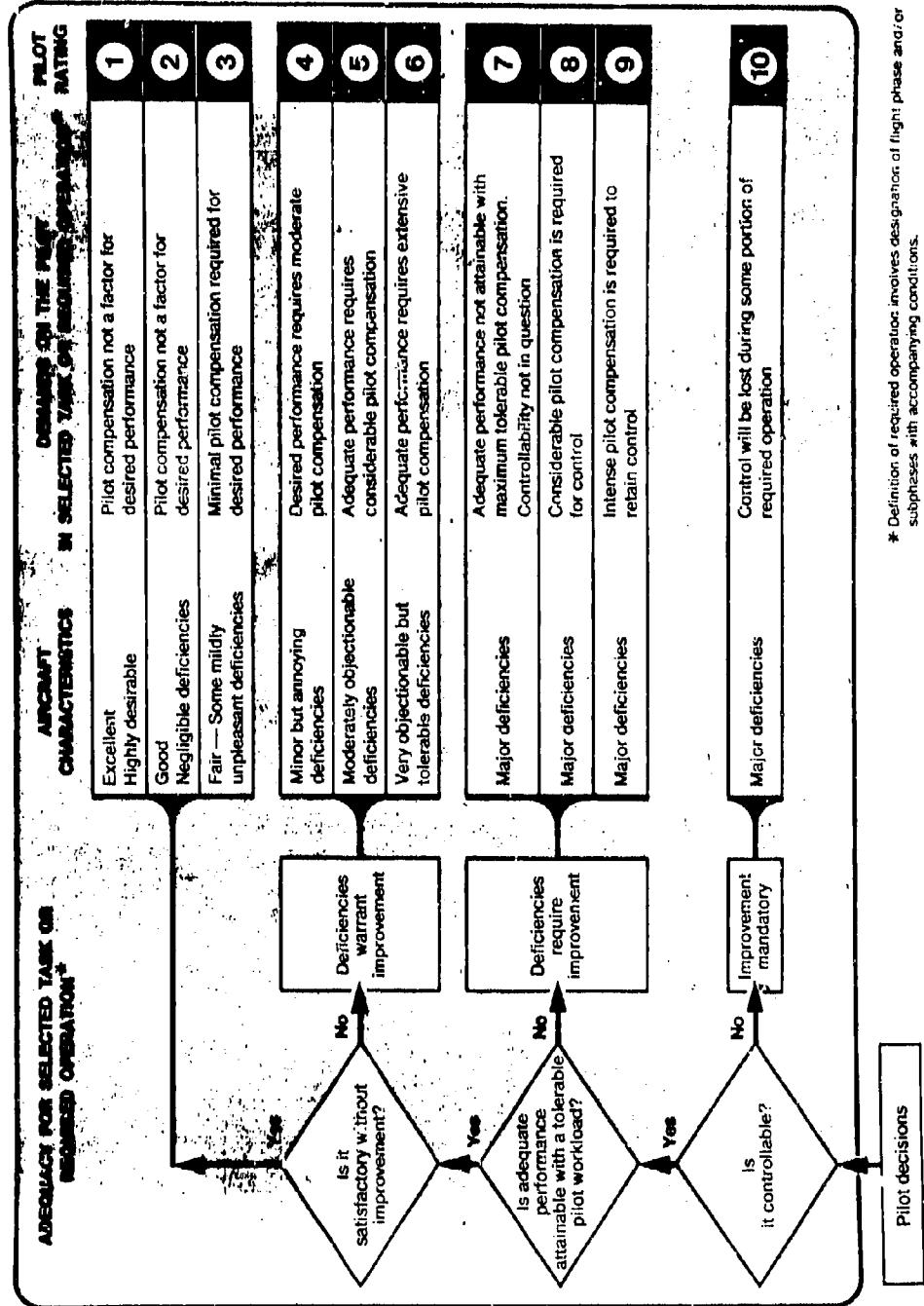
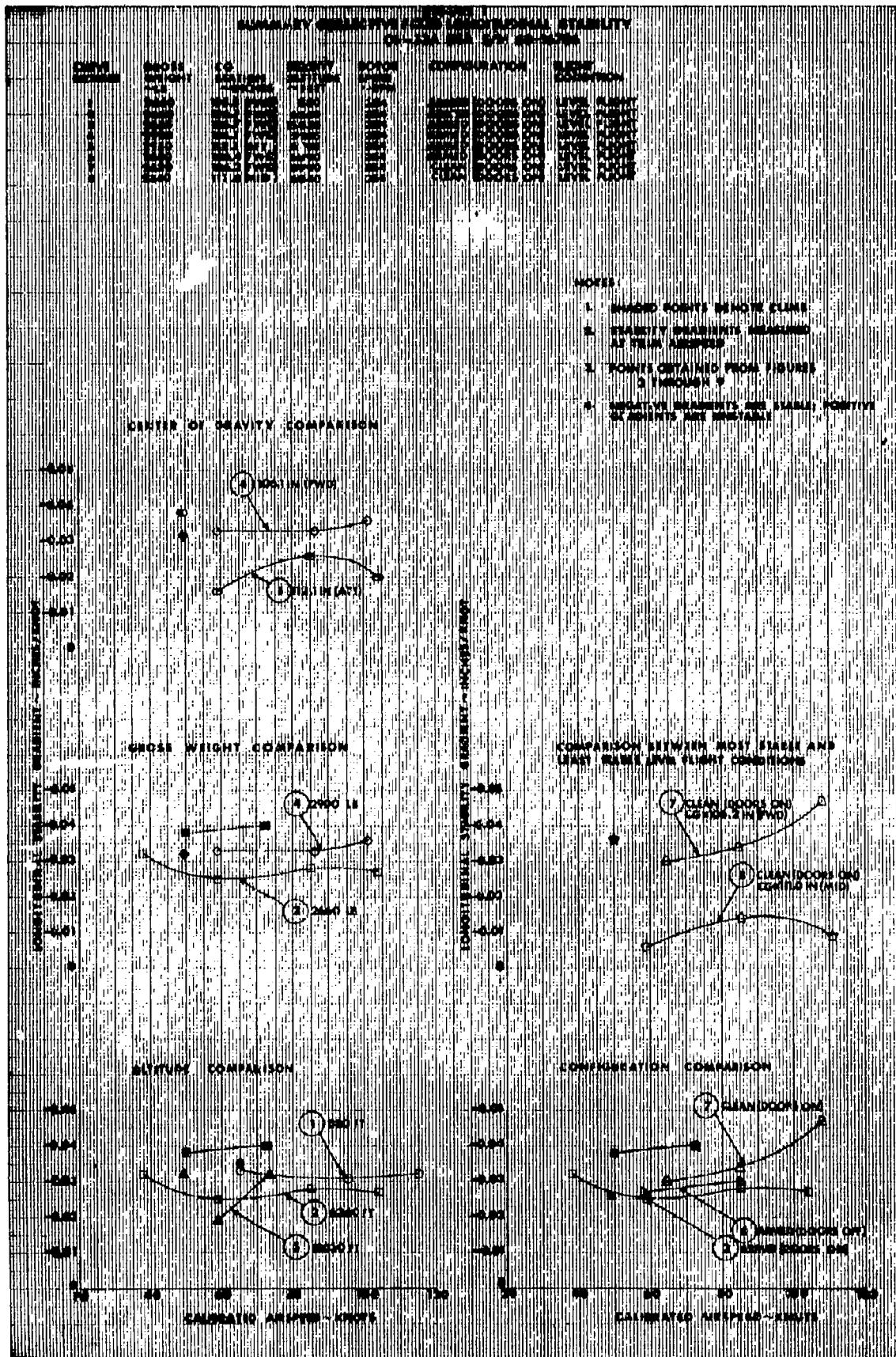


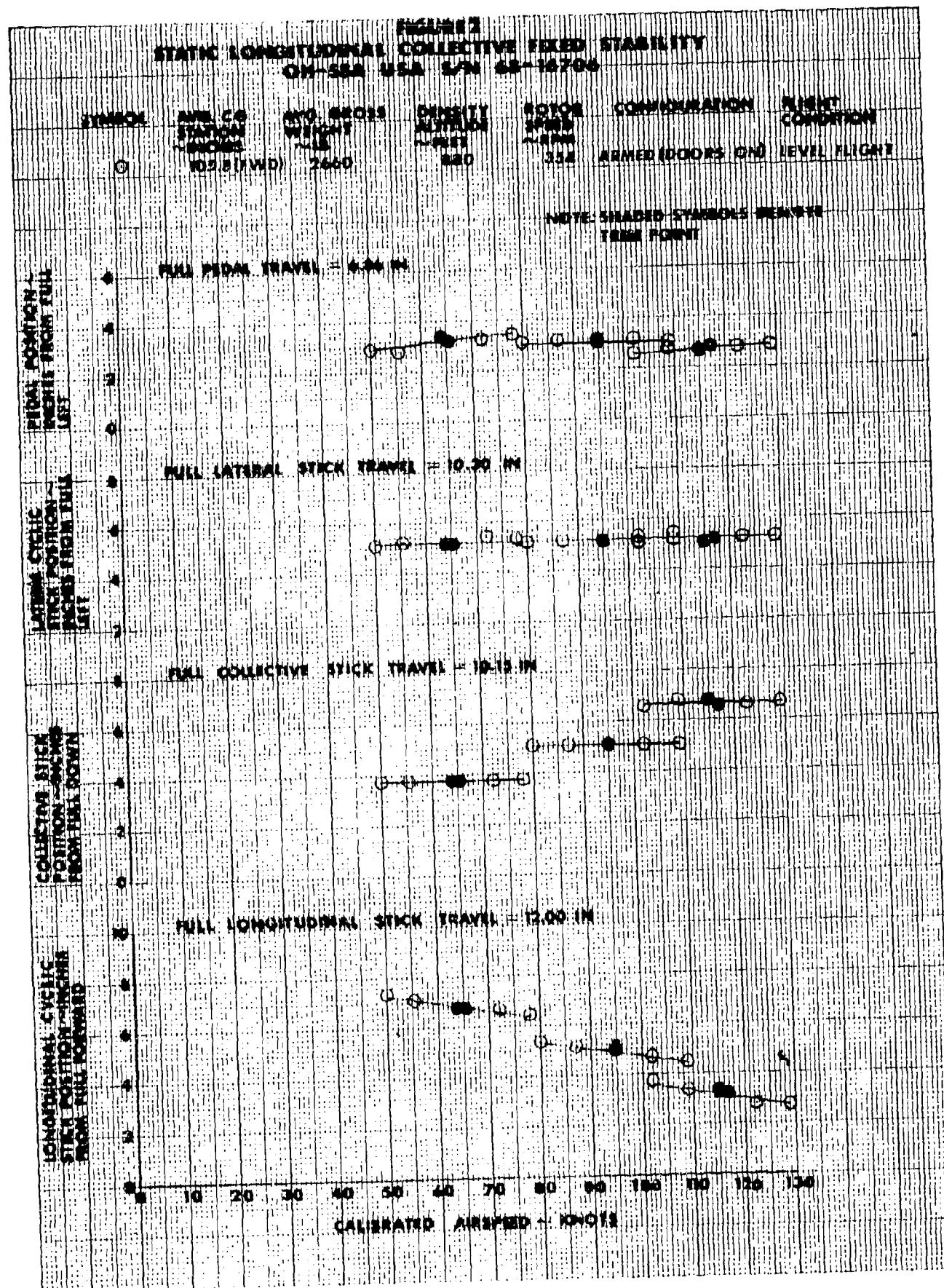
Photo C. Attitude Gyros, Rate Gyros and Angular Accelerometers.

## APPENDIX III. HANDLING QUALITIES RATING SCALE

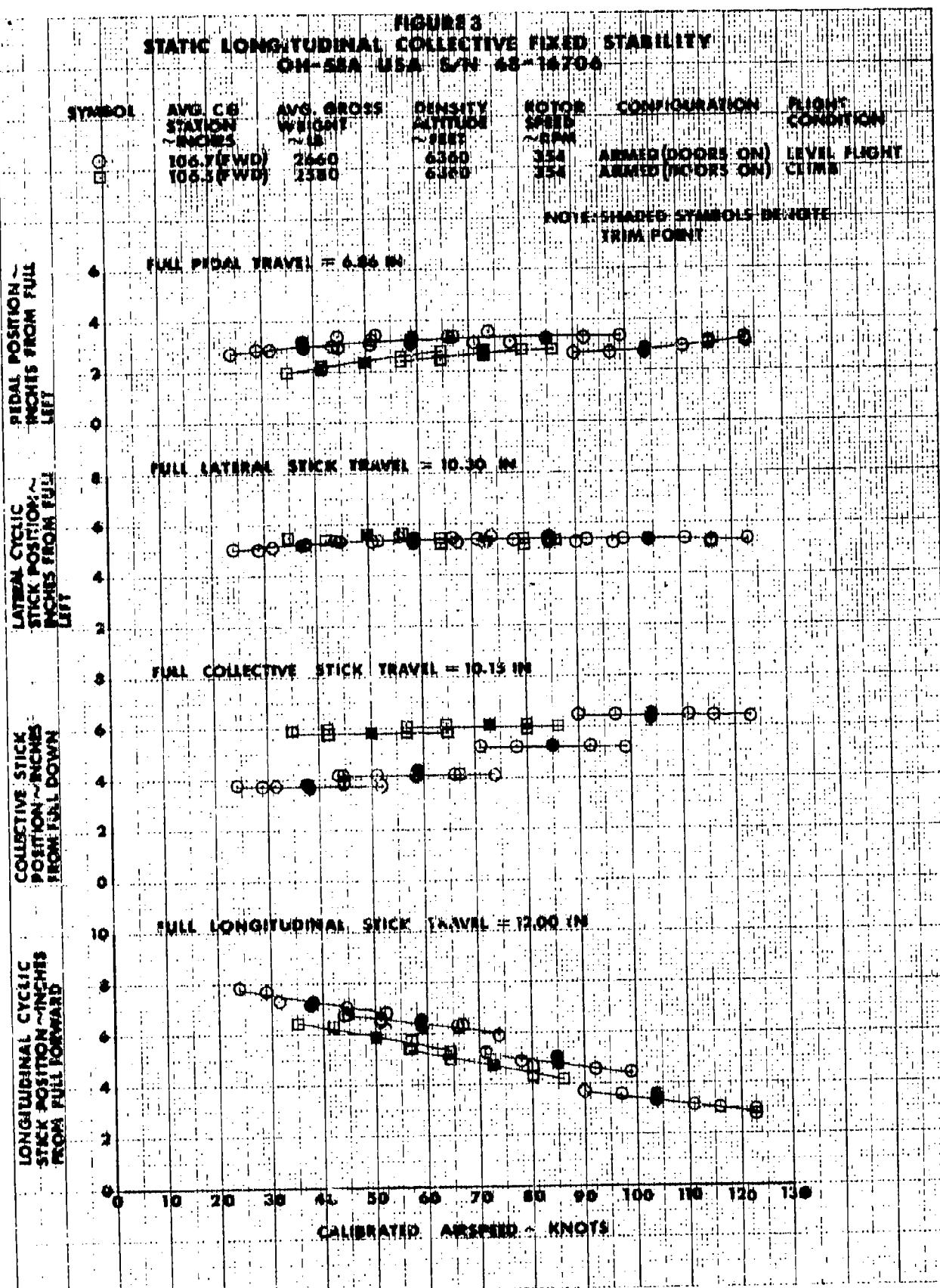


## **APPENDIX IV. TEST DATA**





**FIGURE 3**  
**STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY**  
**OH-58A USA S/N 66-14704**



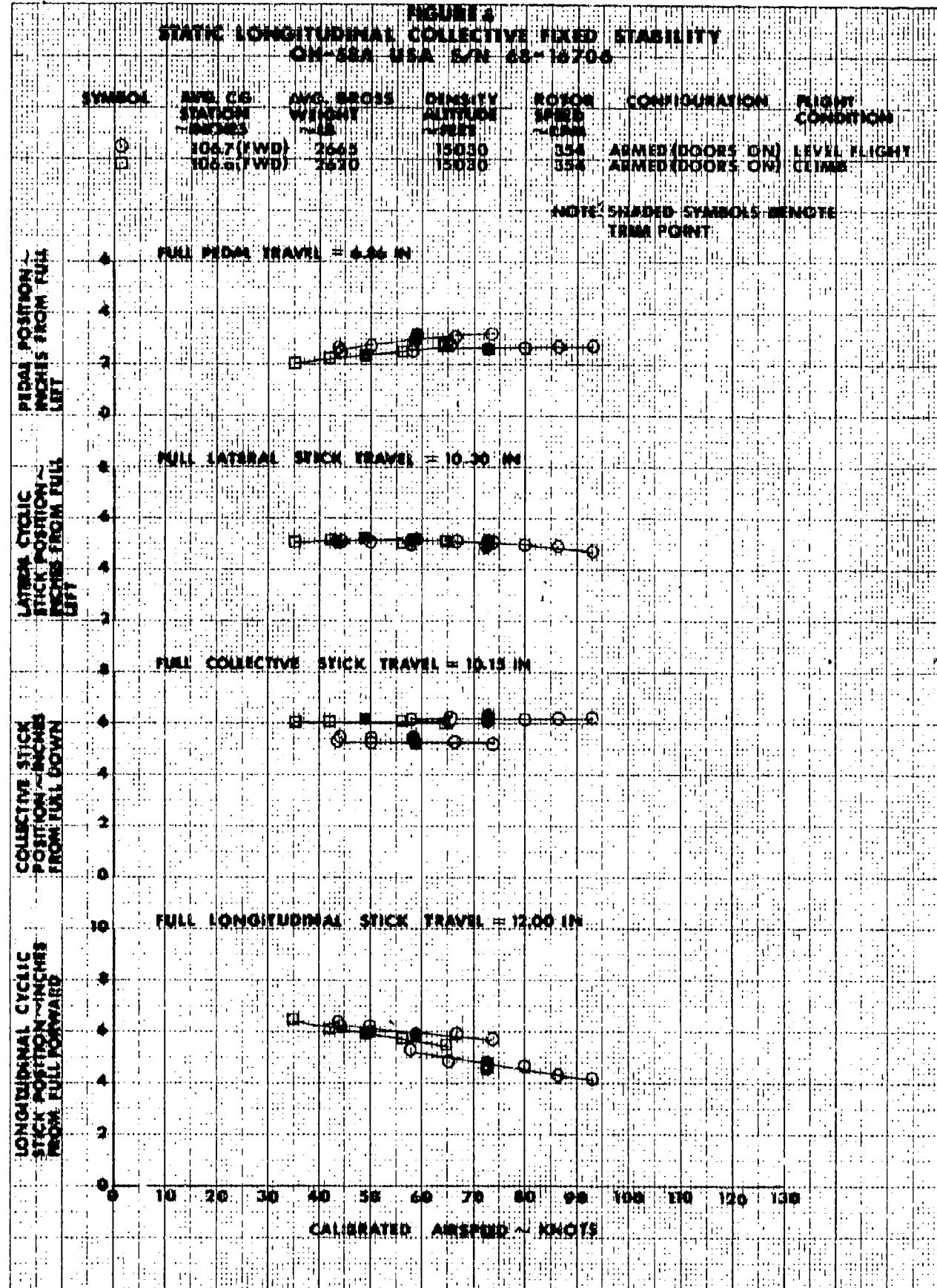


FIGURE 5  
STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY  
OH-58A USA S/N 68-16706

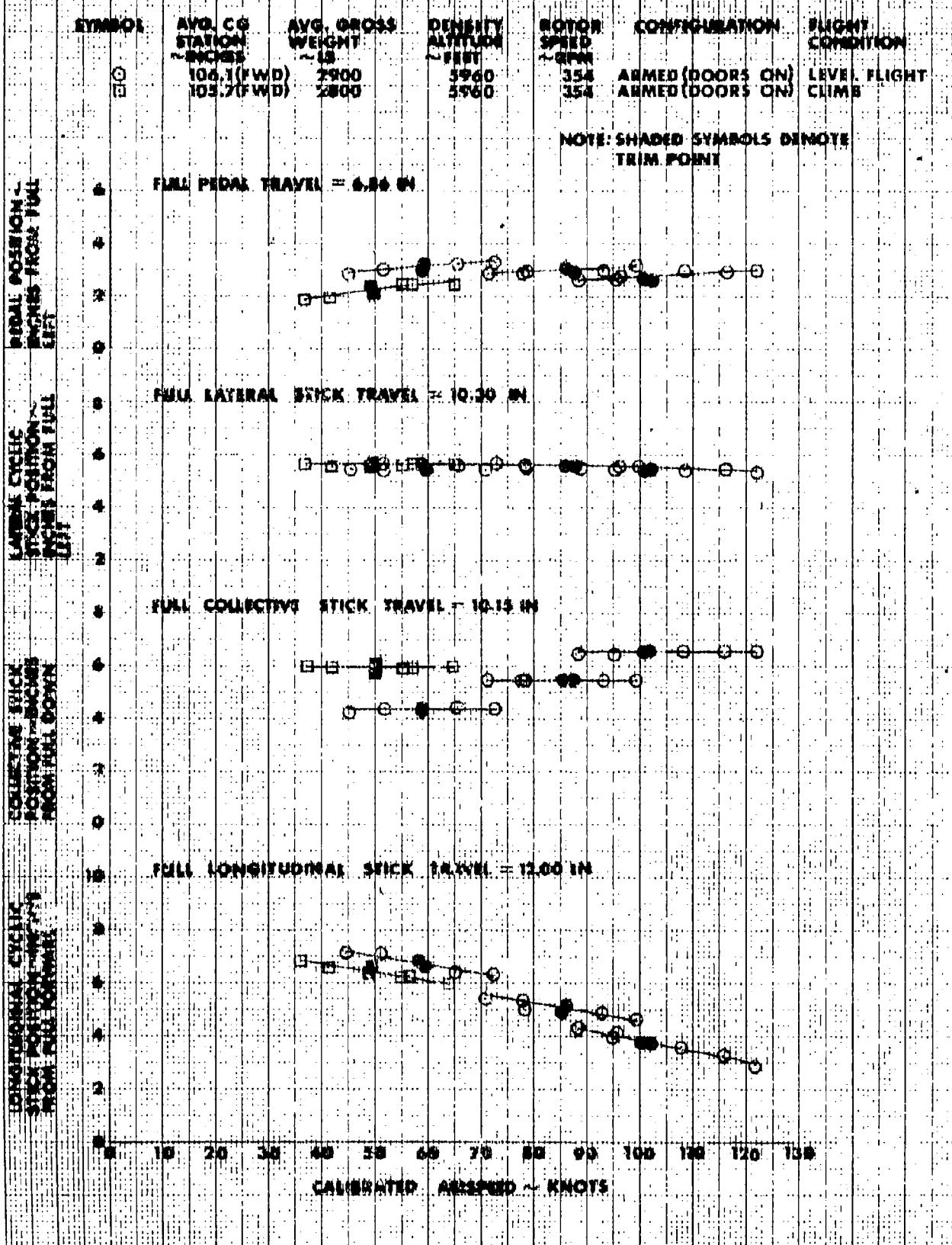
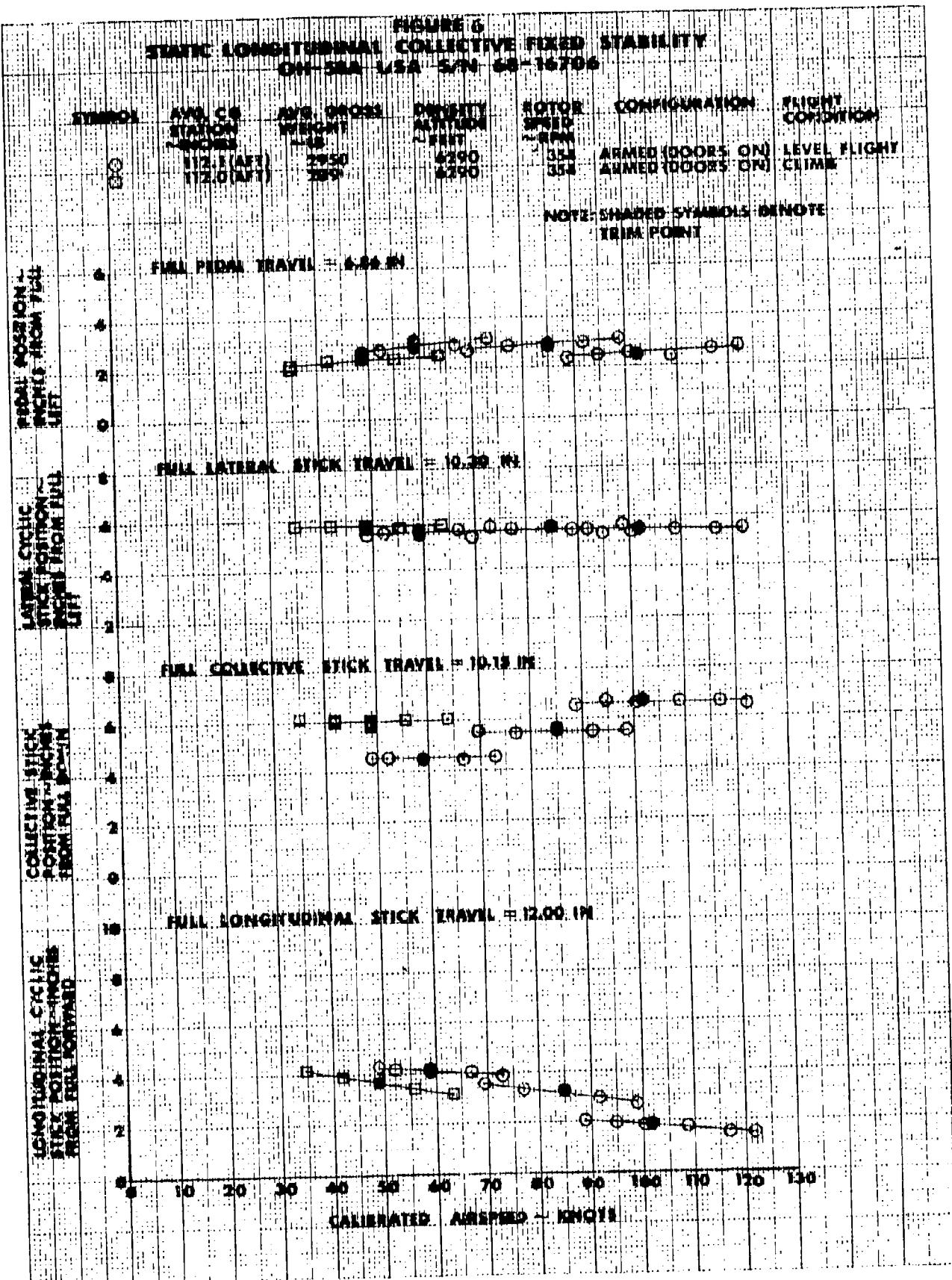
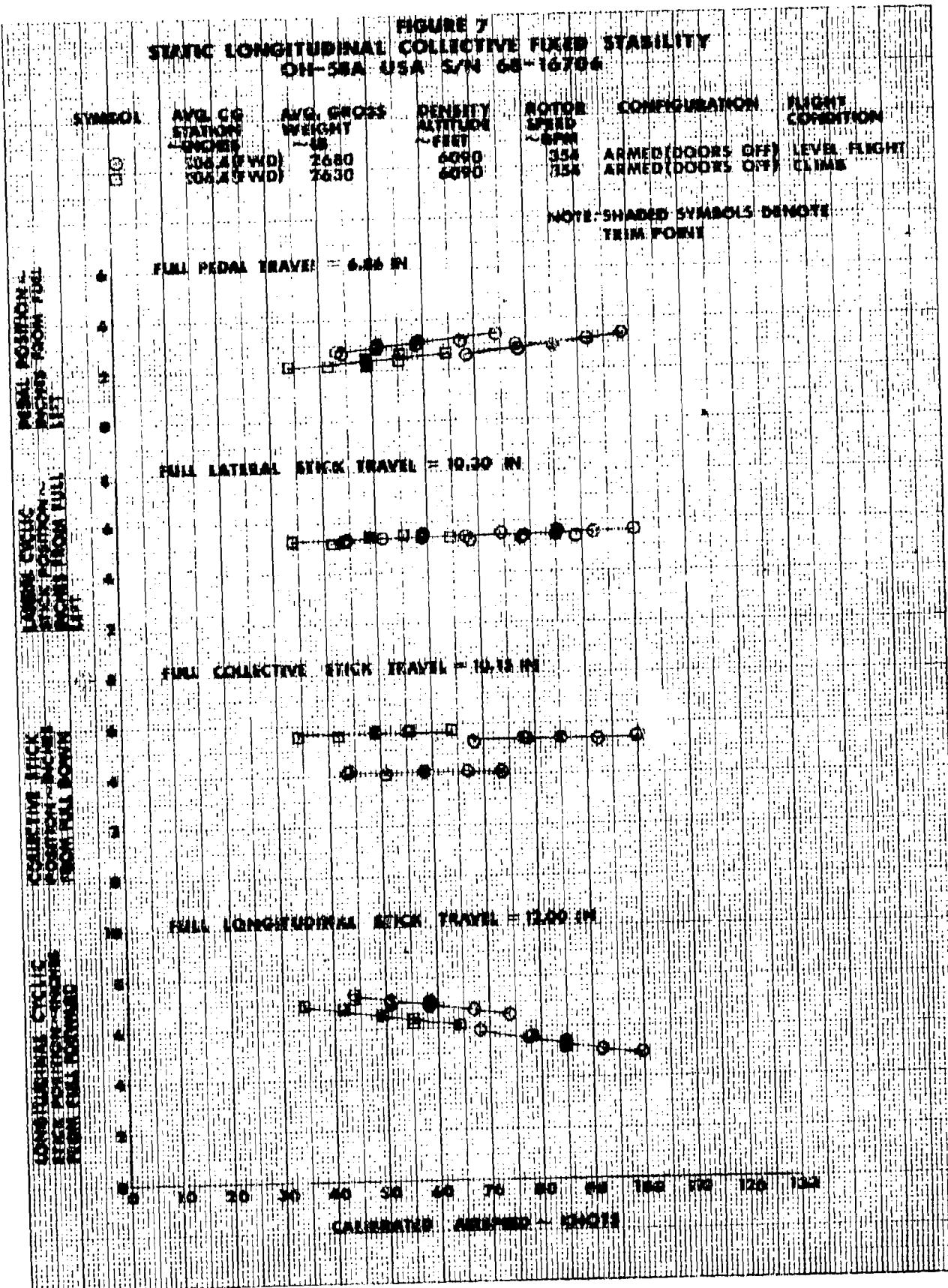


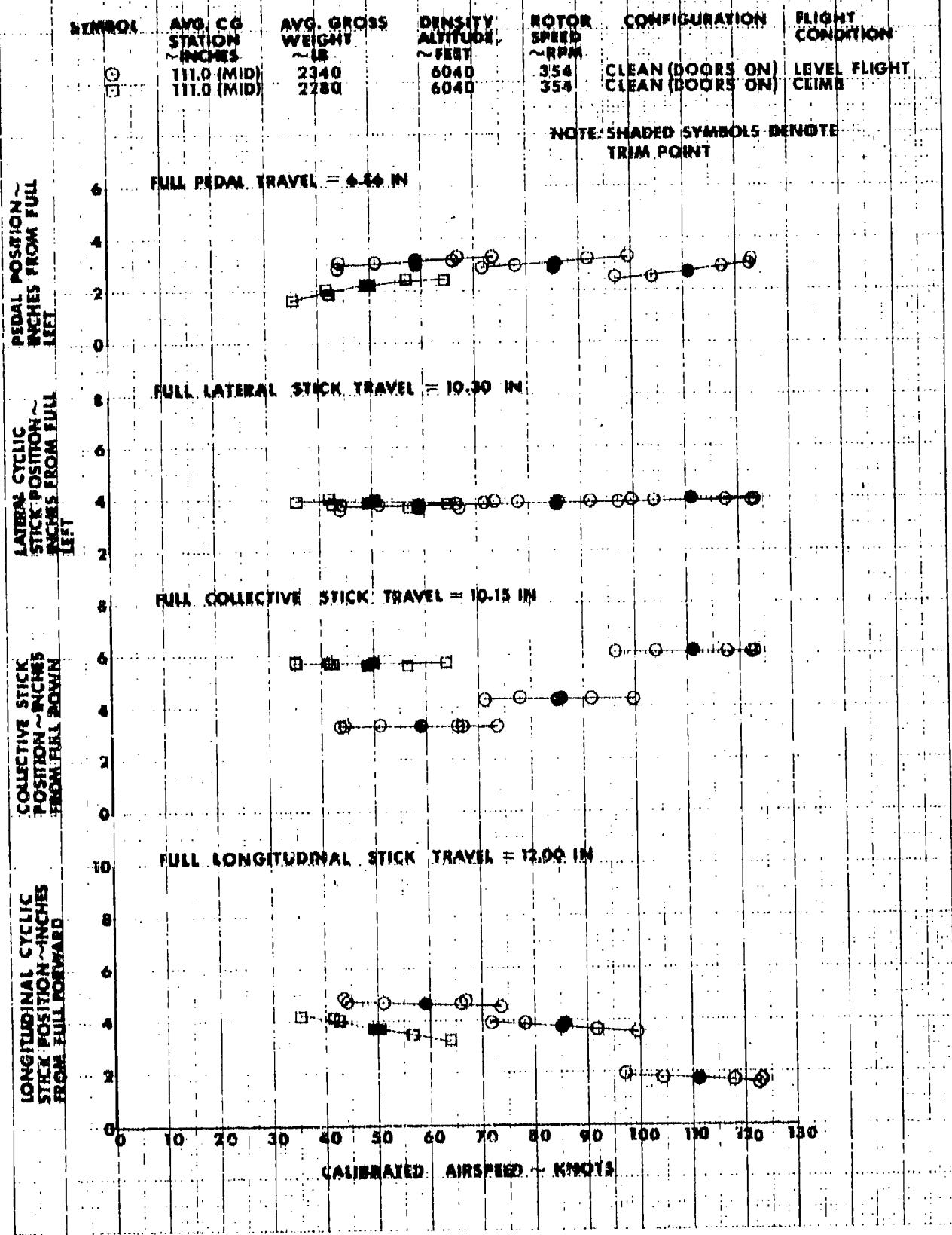
FIGURE 16  
STATIC LOADS THERMAL COLLECTIVE FIXED STABILITY  
OH-58C USA C/N 68-16706



**FIGURE 7**  
**STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY**  
**OH-58A USA S/N 68-16706**



**FIGURE 8**  
**STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY**  
**OH-58A USA S/N 68-16706**

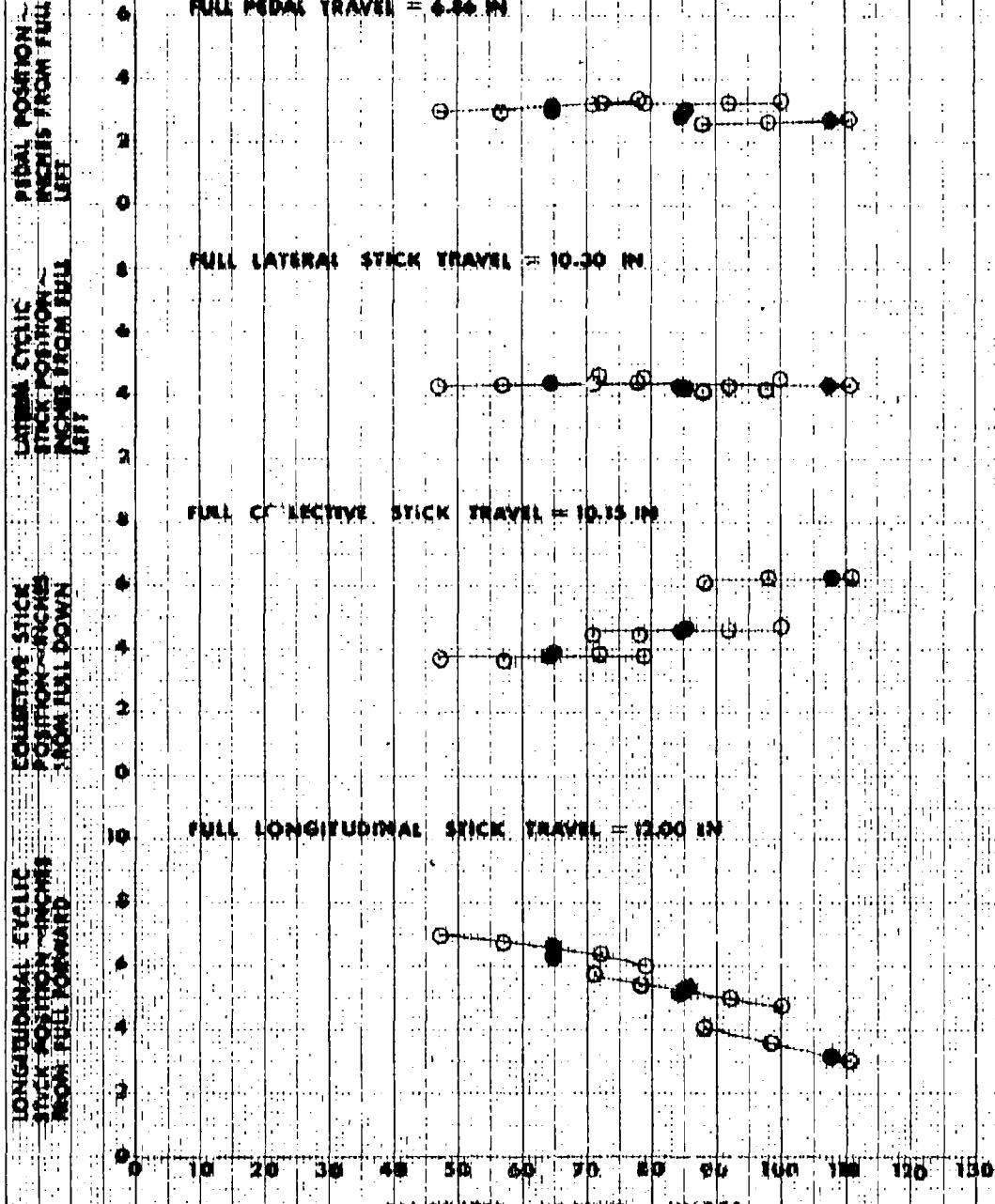


**STATIC LONGITUDINAL COLLECTIVE FIXED STABILITY**  
**OH-58A USA 3/28/68 10706**

SYMBOL	AVG. CO STATION	AVG. GROSS WEIGHT	DENSITY ALTITUDE	MOTOR SPDSES	CONFIGURATION	FLIGHT CONDITION	
③	~MACH3	~14	~FTES	~RPM	354	CLEAN (DOORS ON)	LEVEL FLIGHT
	106.2(FWD)	2620	5990				

NOTE: SHADED SYMBOLS DENOTE  
TRIM POINT

**FULL PEDAL TRAVEL = 6.34 IN.**



CALIBRATED. ACCEPTED ~ KNOTS.

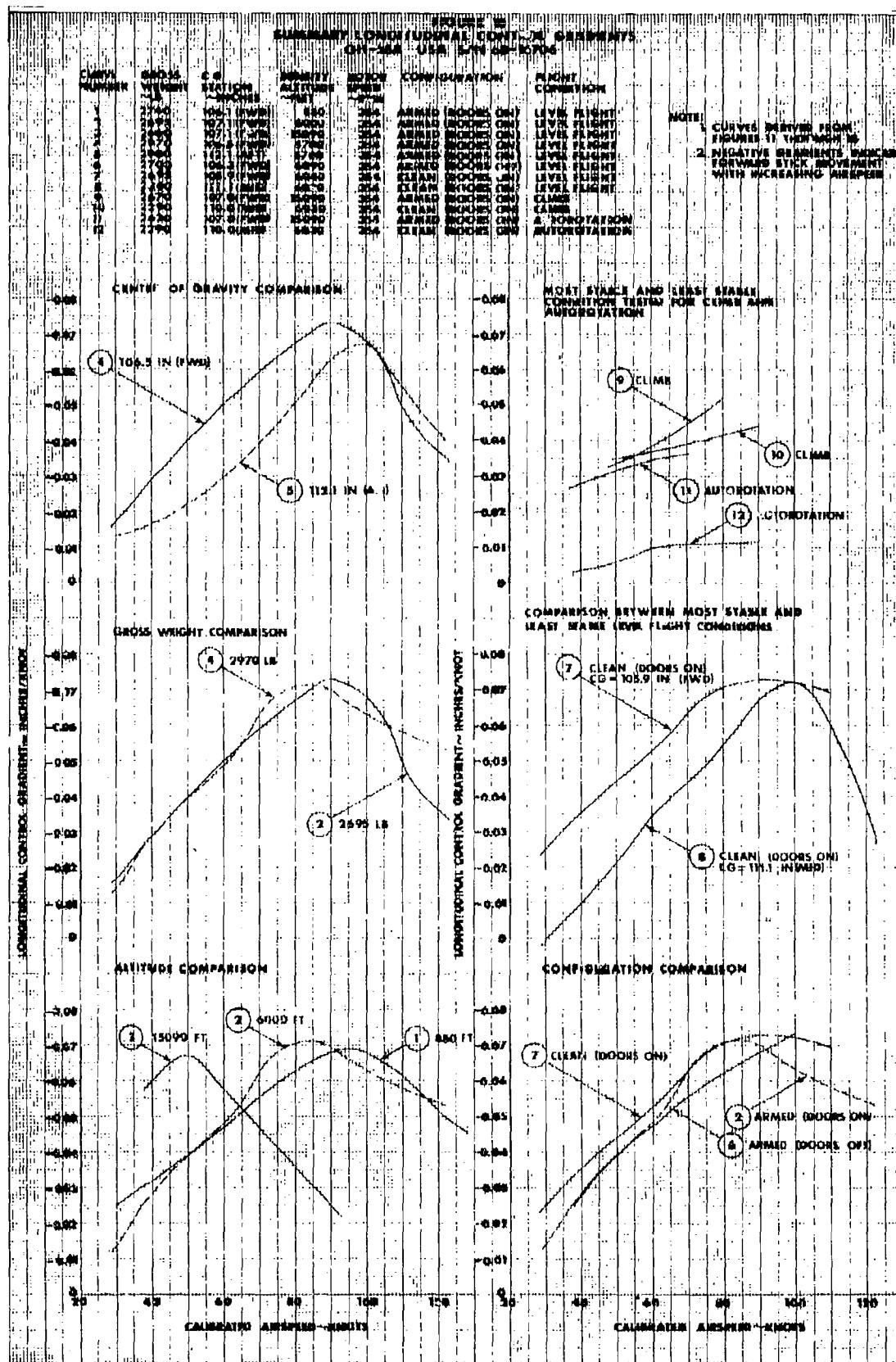
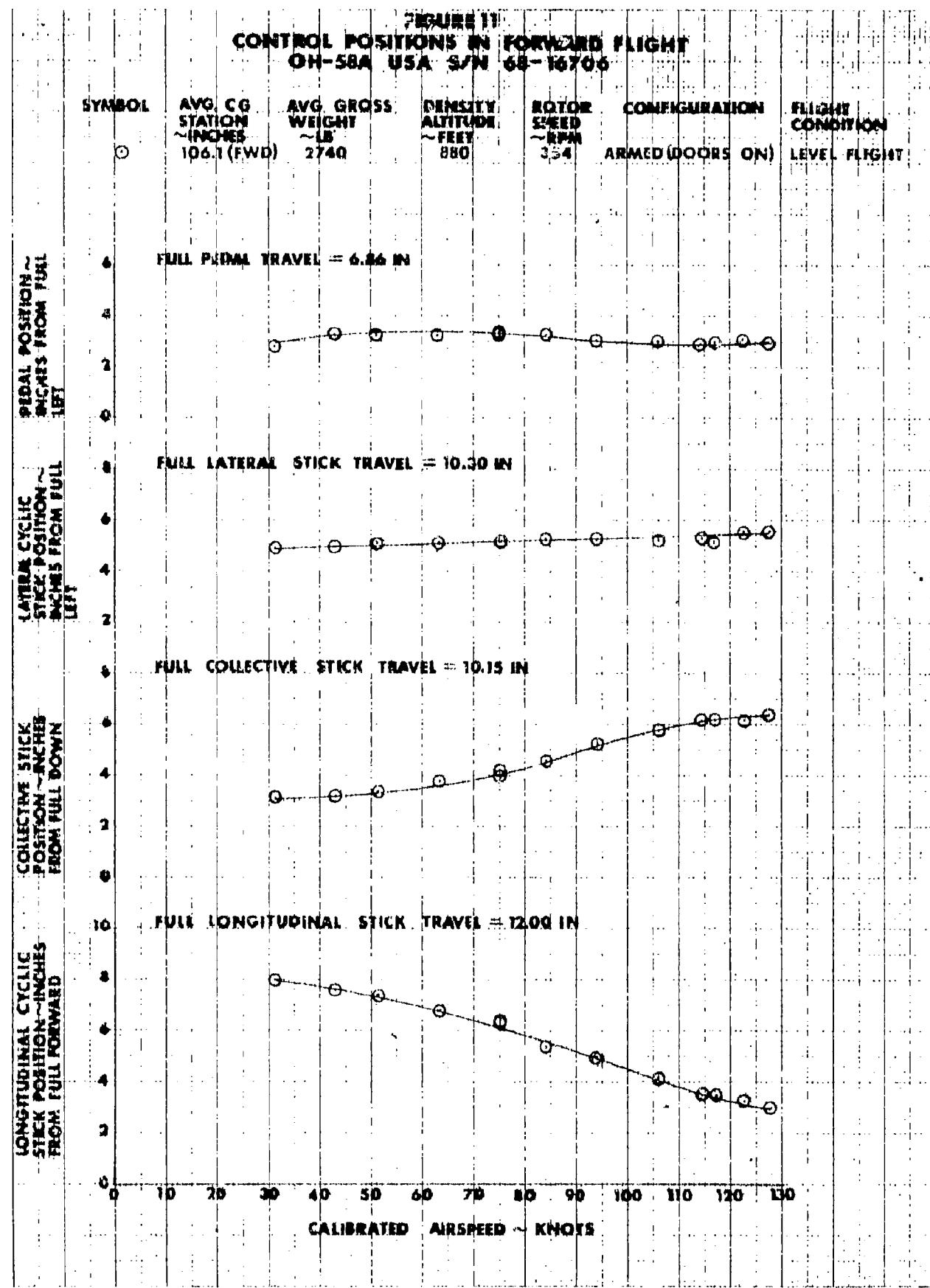


FIGURE 11  
CONTROL POSITIONS IN FORWARD FLIGHT  
OH-58A USA S/N 68-16706



**FIGURE 12**  
**CONTROL POSITIONS IN FORWARD FLIGHT**  
**ON-SITE USA S/N 62-16706**

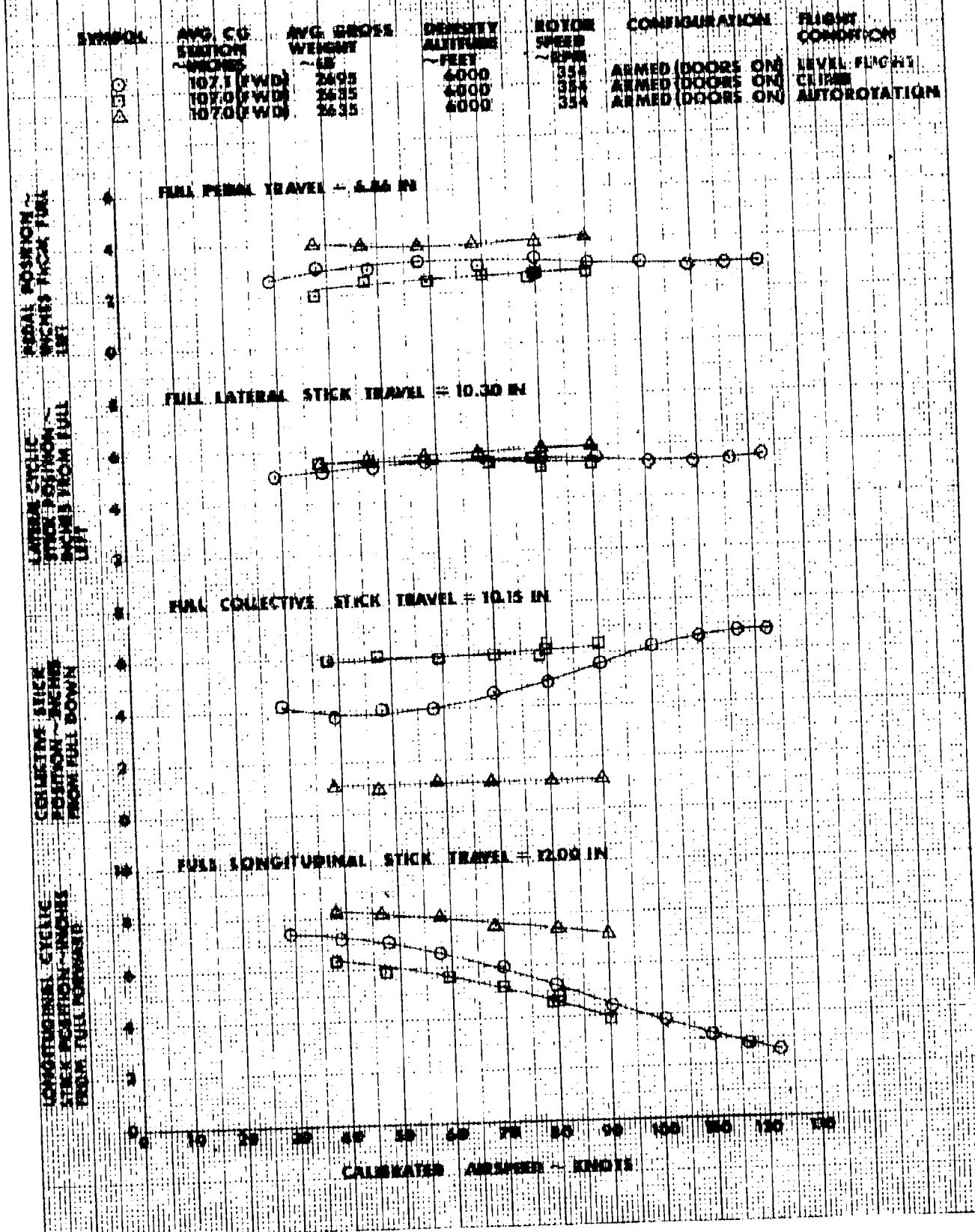


FIGURE 13  
CONTROL POSITIONS IN FORWARD FLIGHT  
ON-SEA USA 1/11 88-16708

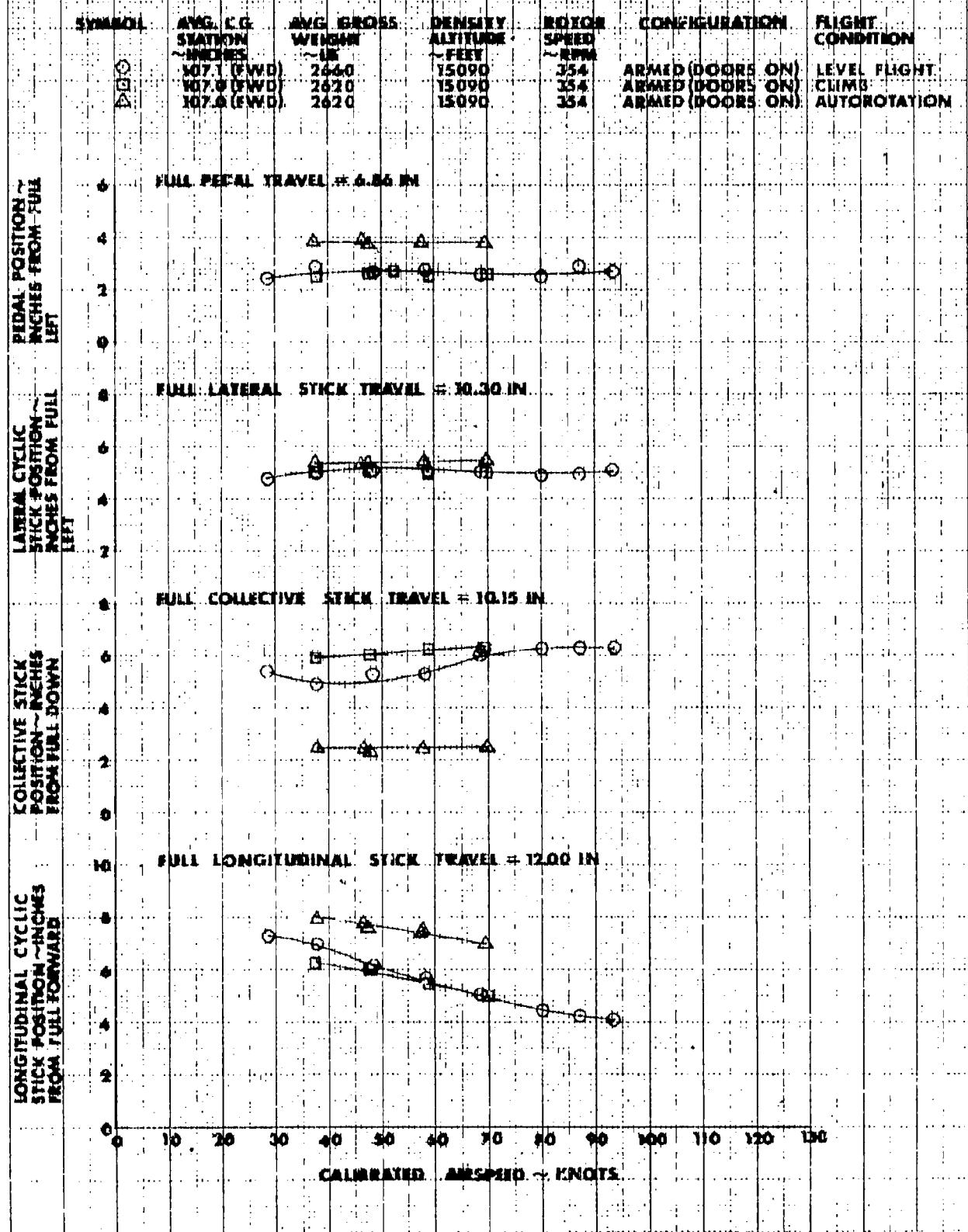
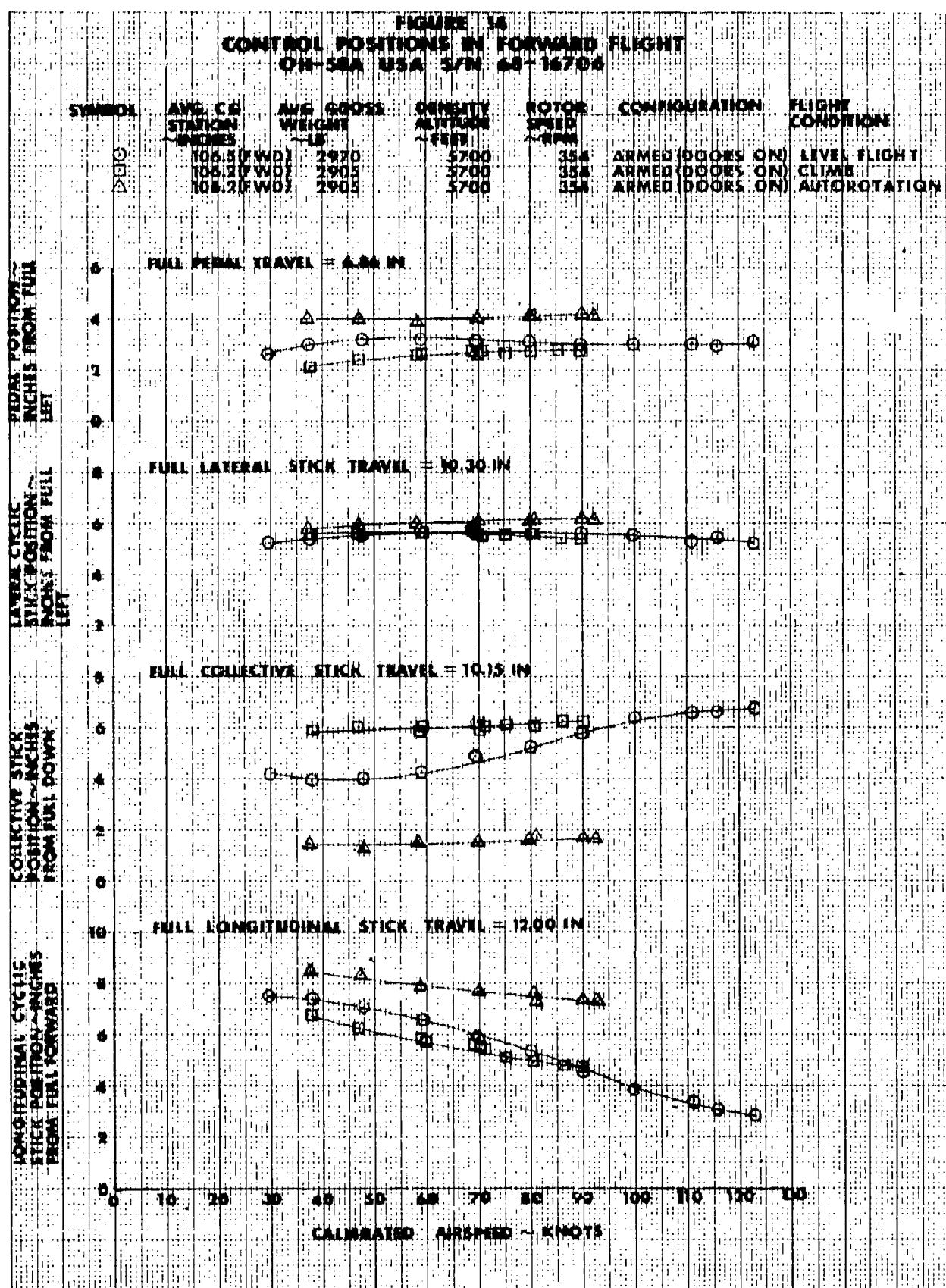


FIGURE 14  
CONTROL POSITIONS IN FORWARD FLIGHT  
OH-58A USA S/N 68-16704



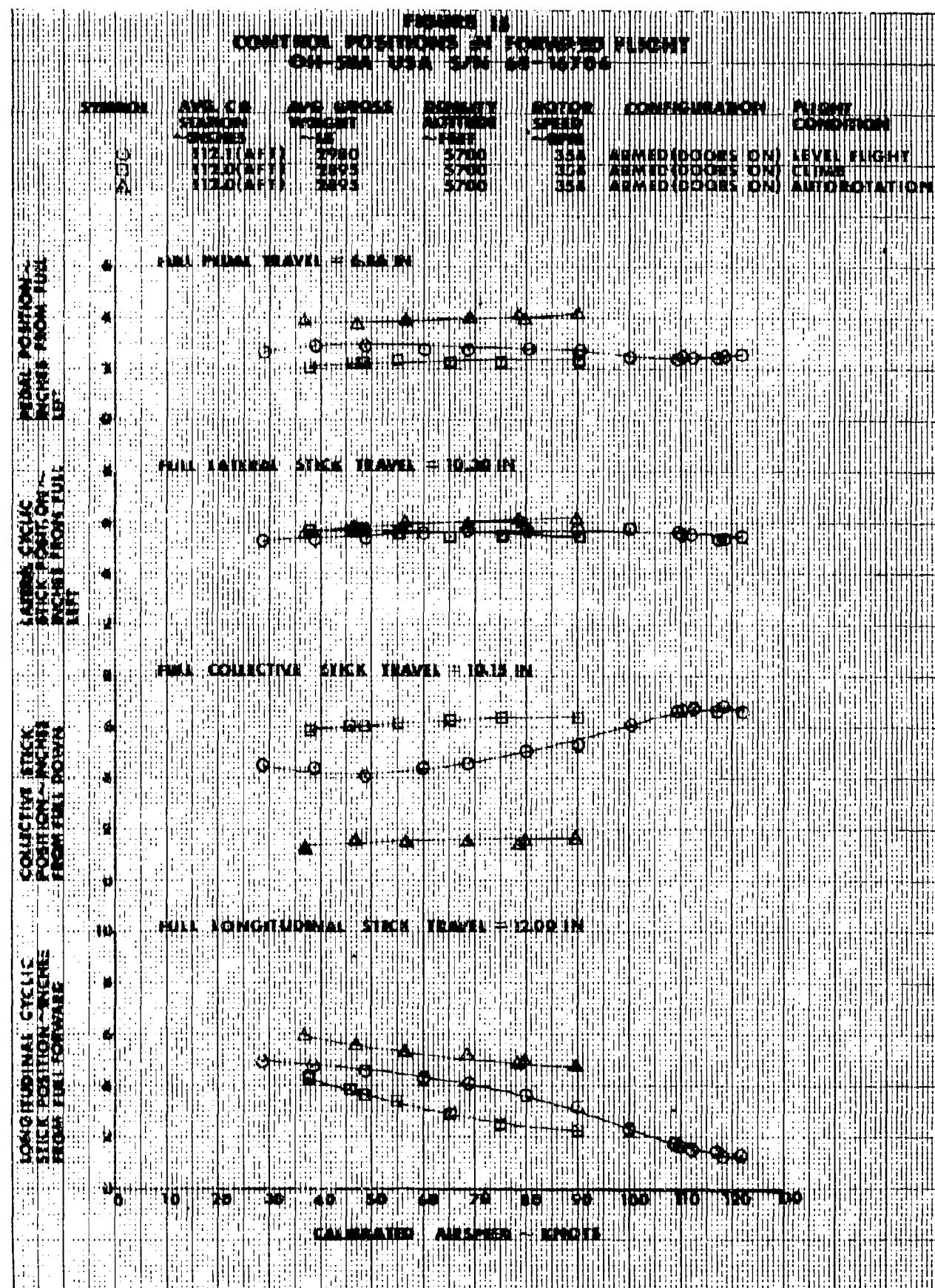


FIGURE 16  
CONTROL POSITIONS IN FORWARD FLIGHT  
ON-SMA USA 3/N 68-16706

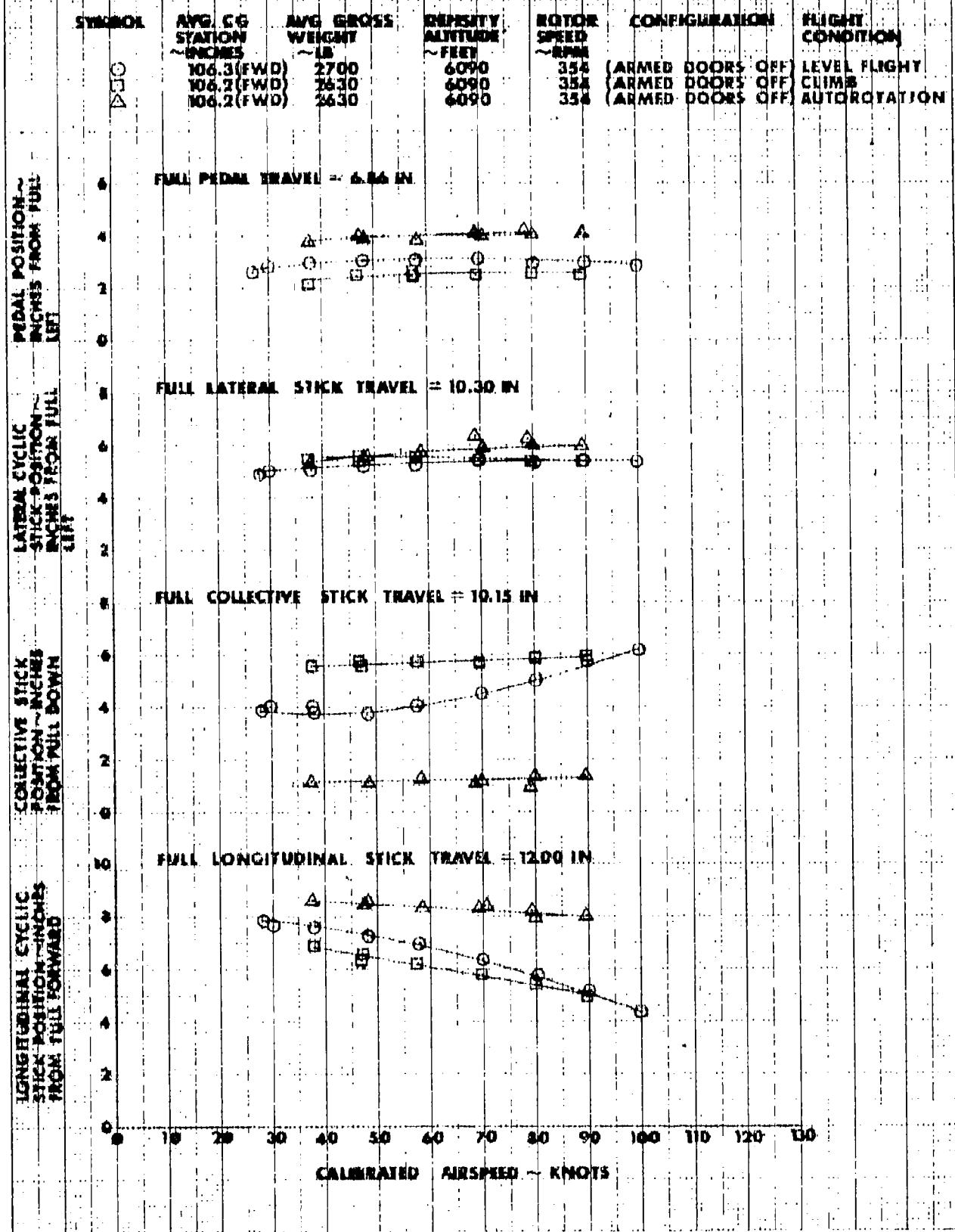


FIGURE 17  
CONTROL POSITIONS IN FORWARD FLIGHT  
OH-SEA USA S/N 68-16706

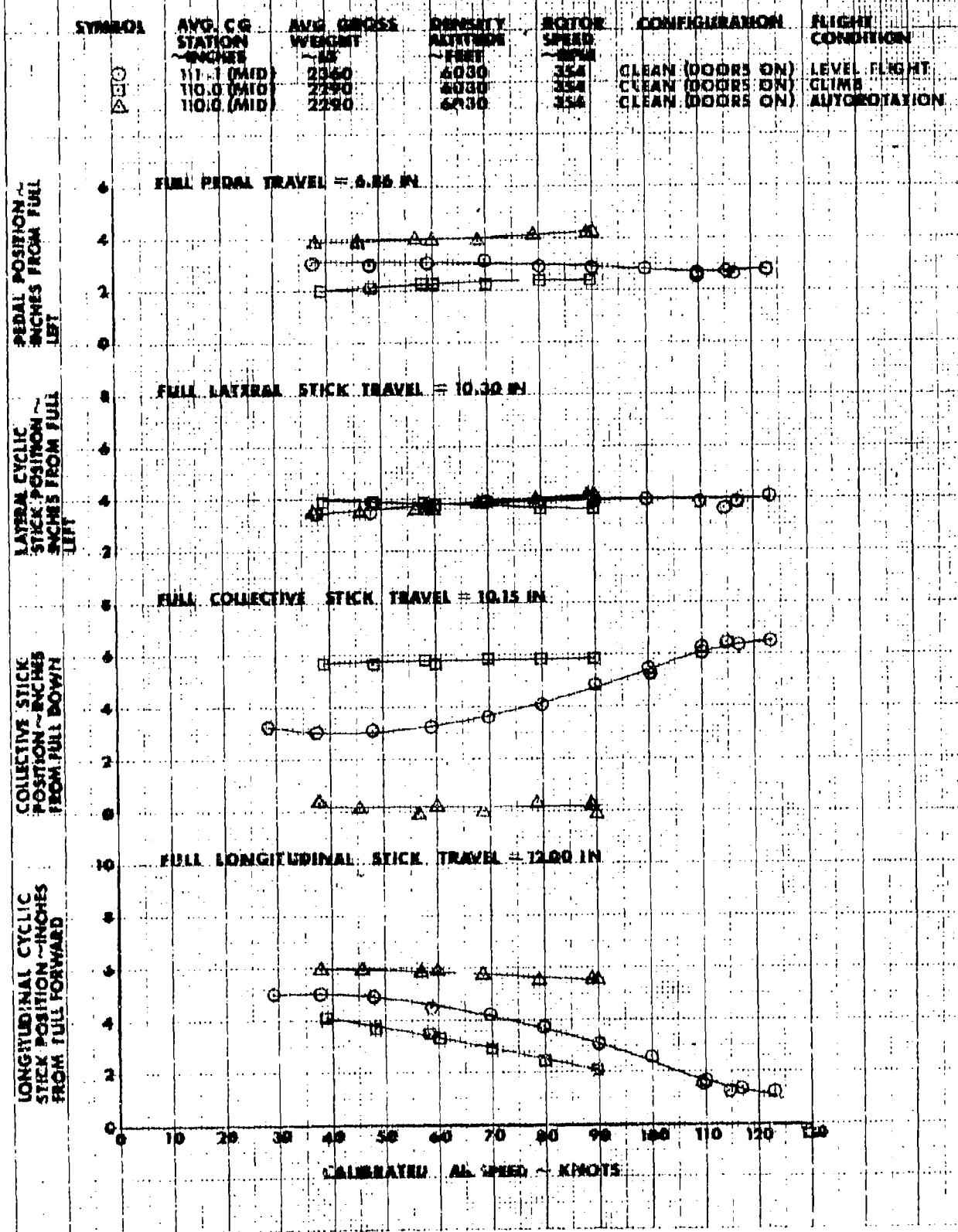
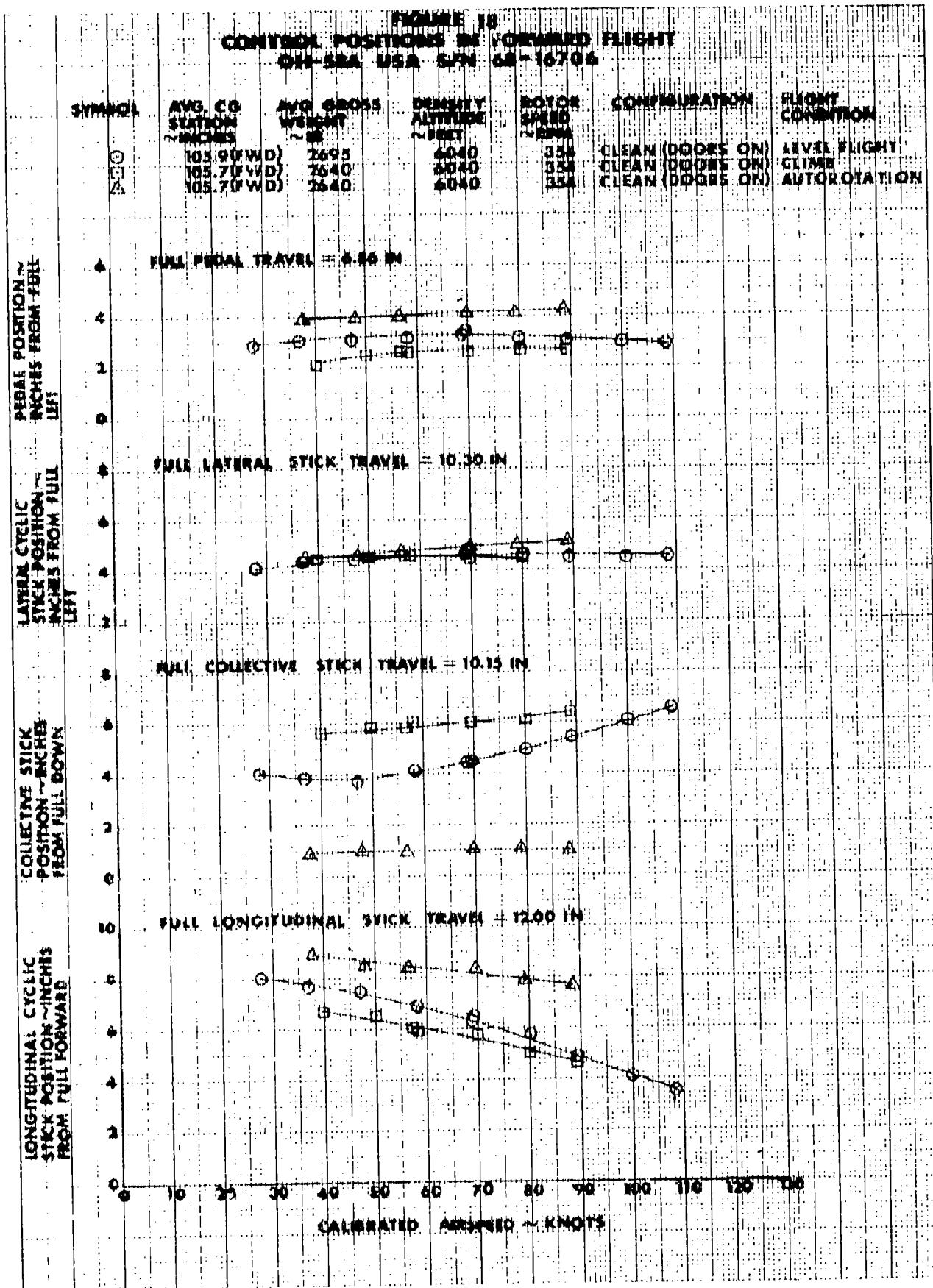


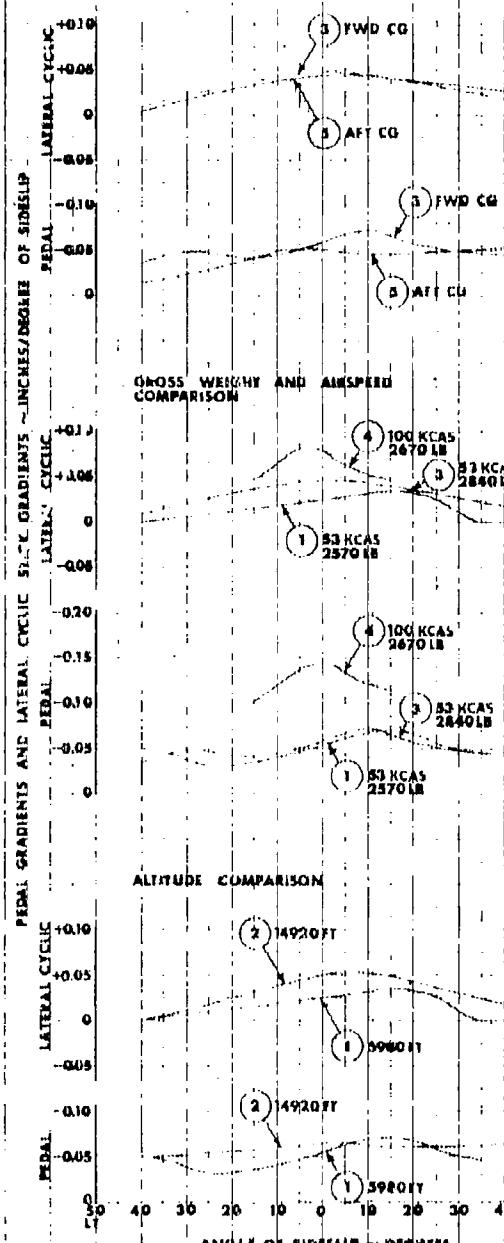
FIGURE 18  
CONTROL POSITIONS IN UNARMED FLIGHT  
OH-58A USA C/N 08-16706



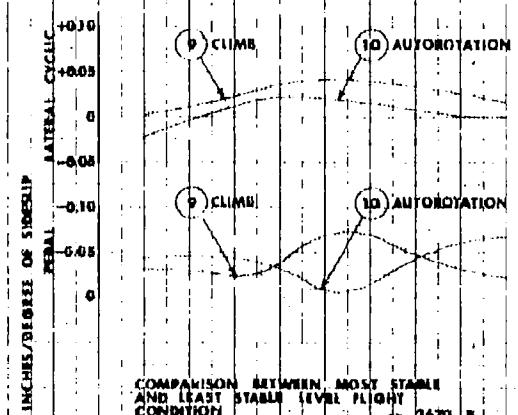
**FIGURE 18  
STATIC LATERAL-DIRECTIONAL STABILITY GRADIENT SUMMARY  
OM-SEA USA S/N 48-14704**

CURVE NUMBER	GROSS WEIGHT ~ LB	C.G. POSITION	DENSITY ALTITUDE	MOTOR RPM	CONFIGURATION	FLIGHT CONDITION	CALIBRATED AIRSPEED - KTS
1	2520	104.8 FWD	5980	354	ARMED DOORS ON	LEVEL FLIGHT	73
2	2670	107.2 FWD	5980	354	ARMED DOORS ON	LEVEL FLIGHT	73
3	2640	107.4 FWD	6020	354	ARMED DOORS ON	LEVEL FLIGHT	73
4	2640	107.1 MID	5980	354	ARMED DOORS ON	LEVEL FLIGHT	73
5	2640	112.1 AFT	5970	354	ARMED DOORS ON	LEVEL FLIGHT	73
6	2640	111.1 MID	5970	354	CLEAN DOORS ON	LEVEL FLIGHT	73
7	2640	107.0 FWD	5970	354	ARMED DOORS ON	LEVEL FLIGHT	73
8	2640	107.0 FWD	5970	354	ARMED DOORS ON	CLIMB	73
9	2640	107.0 FWD	5970	354	ARMED DOORS ON	AUTOROTATION	73
10	2640	107.0 FWD	5970	354	ARMED DOORS ON	NOTE: CURVES DERIVED FROM FIGURES 20 THROUGH 23 AND FIGURES 24 THROUGH 29. 2. POSITIVE LATERAL CYCLIC GRADIENTS ARE STABLE 3. NEGATIVE PEDAL GRADIENTS ARE STABLE	73

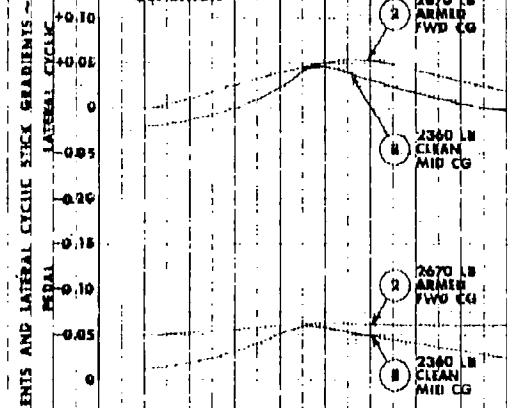
**CENTER OF GRAVITY COMPARISON**



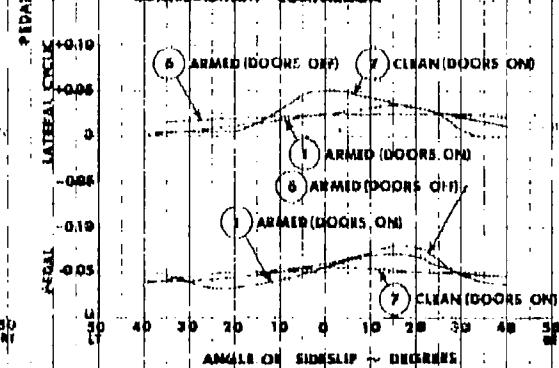
**COMPARISON BETWEEN CLIMB AND AUTOROTATION FLIGHT CONDITIONS**



**COMPARISON BETWEEN MOST STABLE AND LEAST STABLE LEVEL FLIGHT CONDITION**



**CONFIGURATION COMPARISON**



**FIGURE 20**  
**STATIC LATERAL DIRECTIONAL STABILITY**  
**OH-58A USA S/N 68-16704**

SYNTH	AVG. C.G. STATION ~INCHES	AVG. GROSS WEIGHT ~LB.	DENSITY ALTITUDE ~FEET	ROTOR SPEED ~RPM	CONFIGURATION	FLIGHT COND	CABIN A/S ~KNOTS
(1)	105.8 (FWD)	2645	16-60	334	ARMED (DOORS ON)	LEVEL	FLIGHT 60
(2)	105.9 (FWD)	2775	16-60	334	ARMED (DOORS ON)	LEVEL	FLIGHT 93
(3)	105.9 (FWD)	2585	16-60	334	ARMED (DOORS ON)	LEVEL	FLIGHT 113

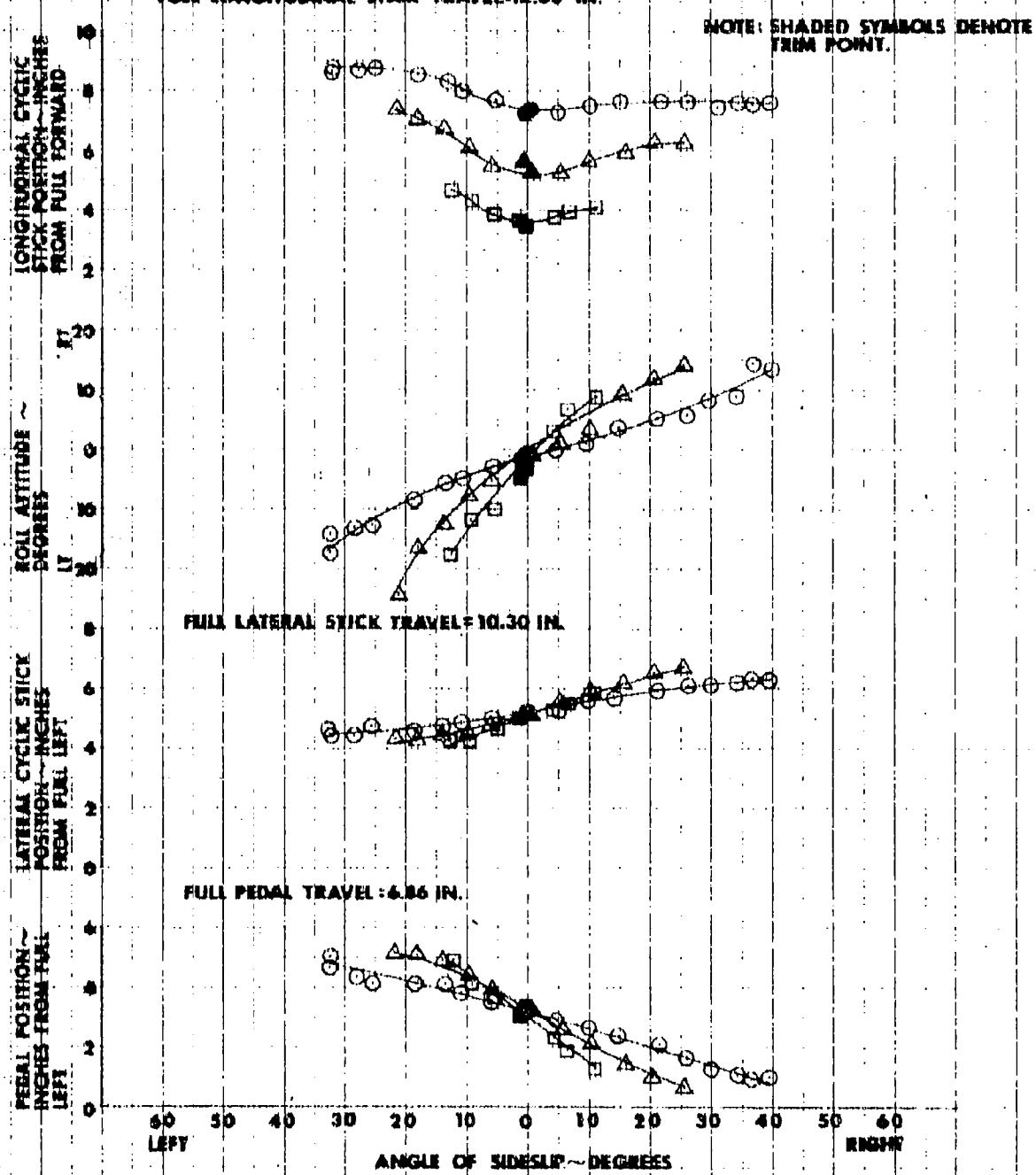
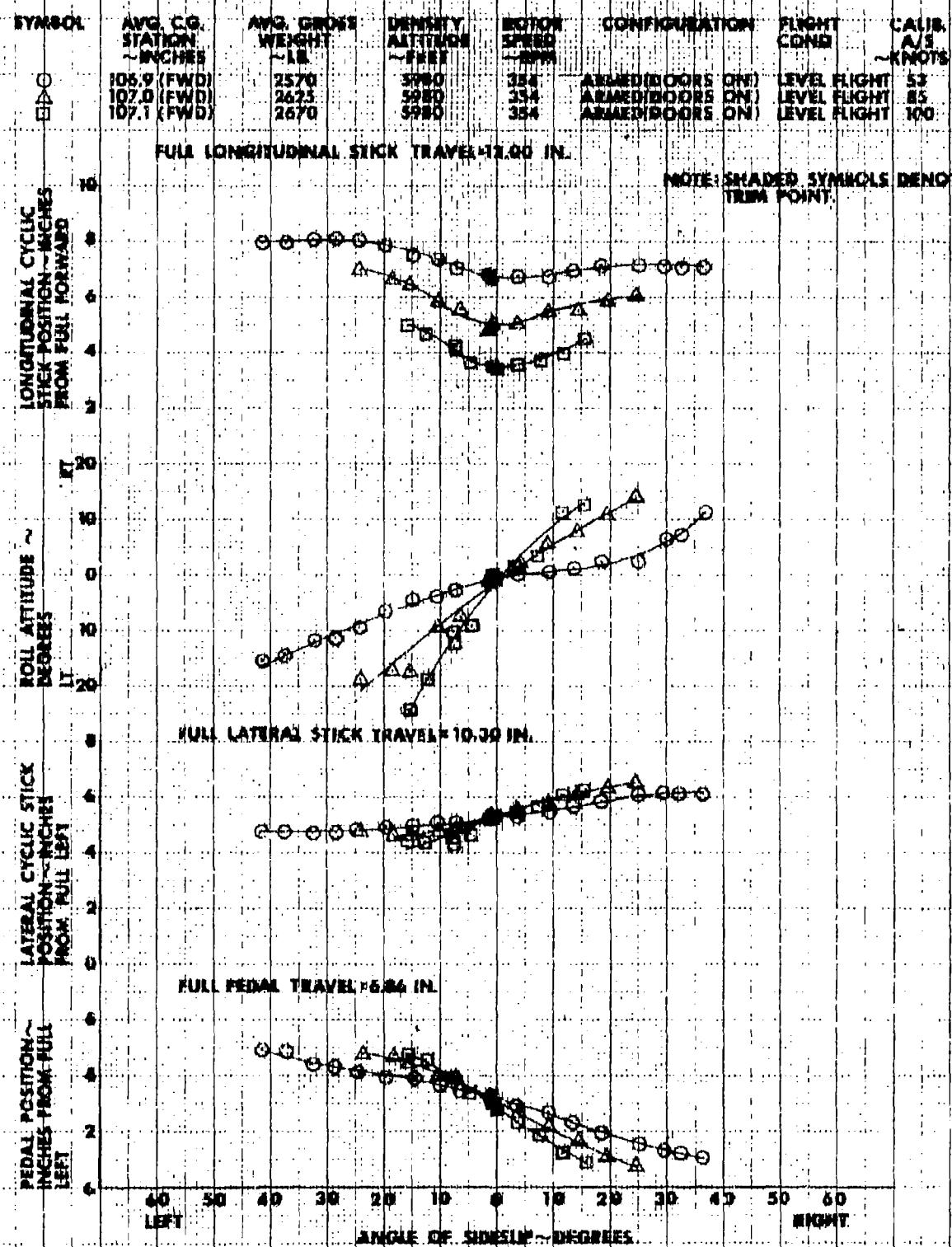


FIGURE 21  
STATIC LATERAL DIRECTIONAL STABILITY  
OH-3B USA S/N 66-10700

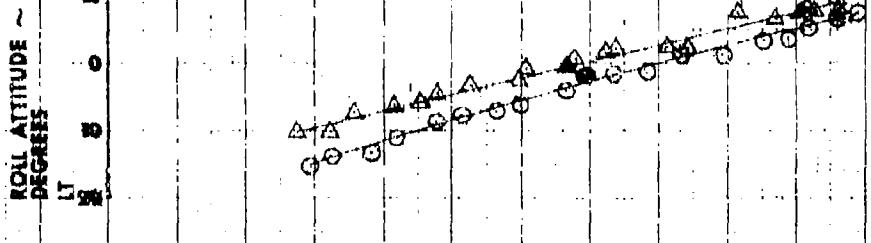
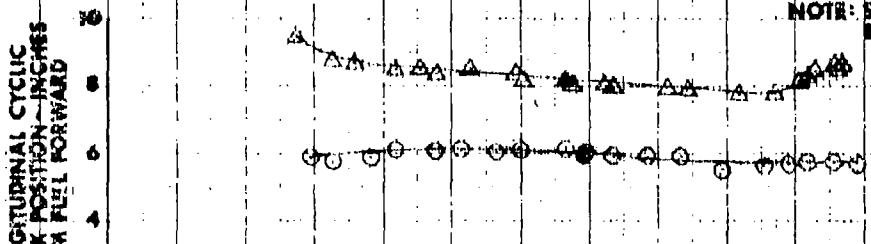


**FIGURE 22**  
**STATIC LATERAL DIRECTIONAL STABILITY**  
**OH-58D USA S/N 68-16700**

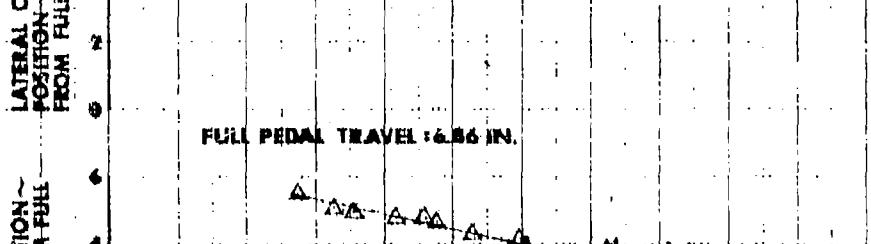
SYMBOL	Avg. CRG STATION ~ INCHES	Avg. GROSS WEIGHT ~ LB	ALTITUDE ~ FEET	ROTOD. SPEED ~ RPM	CONFIGURATION	FLIGHT COND.	CALIB. A/S KNOTS
△	107.0 (FWD)	2620	5790	354	ARMED (DOORS ON)	AUTO	49
○	107.0 (FWD)	2620	5790	354	ARMED (DOORS ON)	CLIMB	49

FULL LONGITUDINAL STICK TRAVEL: 12.00 IN.

NOTE: SHADeD SYMBOLS DENOTE  
TRIM POINT.



FULL LATERAL STICK TRAVEL: 10.30 IN.



FULL PEDAL TRAVEL: 6.86 IN.

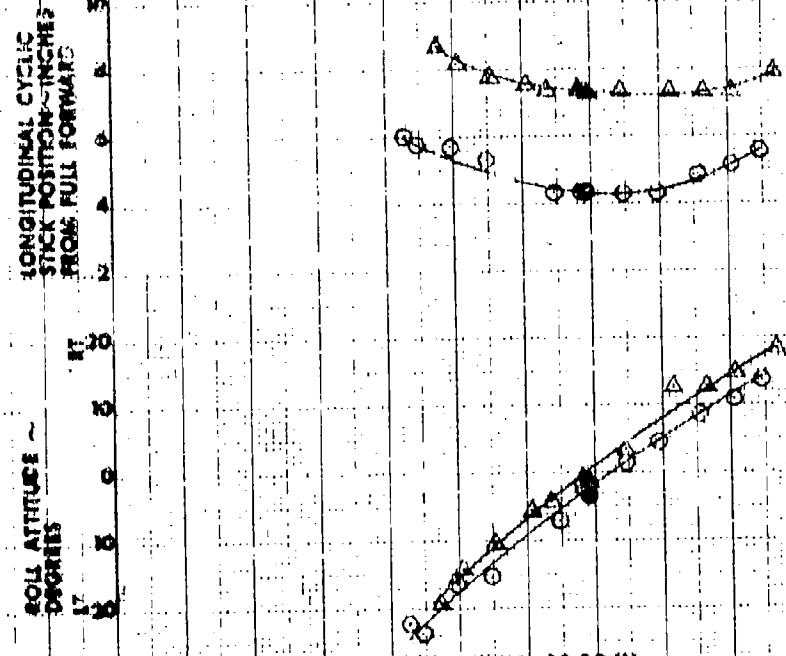
ANGLE OF SIDESLIP ~ DEGREES

**FIGURE 23**  
**STATIC LATERAL DIRECTIONAL STABILITY**  
**FH-SKA USA S/N 68-16706**

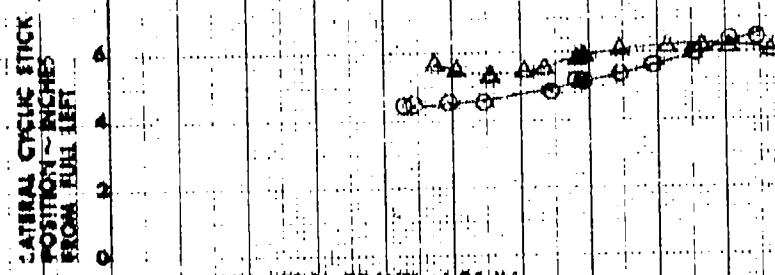
SYMBOL	AVG. CG. STATION ~INCHES	AVG. GROSS WEIGHT ~LB.	DENSITY ALTITUDE ~FEET	ROTOR SPEED ~RPM	COMPOUNDED?	PLANE COND.	CABIN A/S ~KNOTS
	105.9 (FWD)	2665	2000	354	ARMED (DOORS ON)	CLIMB	88
	106.7 (FWD)	2590	5798	354	ARMED (DOORS ON)	AUTO	88

FULL LONGITUDINAL STICK TRAVEL = 12.00 IN.

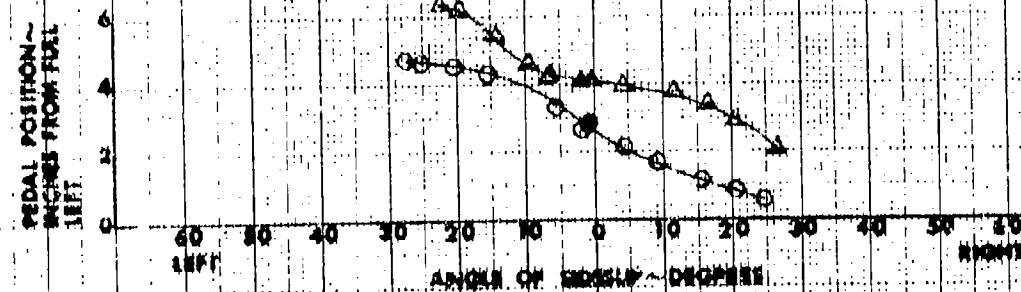
NOTE - SHADeD SYMBOLS DENOTE  
THIN POINT



FULL LATERAL STICK TRAVEL = 10.30 IN.



FULL PEDAL TRAVEL = 6.66 IN.



**FIGURE 24**  
**STATIC LATERAL DIRECTIONAL STABILITY**  
**CH-34A USA S/N 68-16706**

SYMBOL	Avg. CG STATION ~INCHES	Avg. Gross WEIGHT ~LB.	MENITRY ALTITUDE ~FEET	WIND SPEED ~KPH	CONFIGURATION:	FLIGHT COND.	CALC. A/S ~KNOTS
107.2 (FWD)	2670	14920	354	ARMED(DOORS ON)	LEVEL FLIGHT	53	
107.0 (FWD)	2623	14920	354	ARMED(DOORS ON)	LEVEL FLIGHT	74	

FULL LONGITUDINAL STICK TRAVEL=12.00 IN.

NOTE: SHADeD SYMBOLS DENOTE  
TRIM POINT.

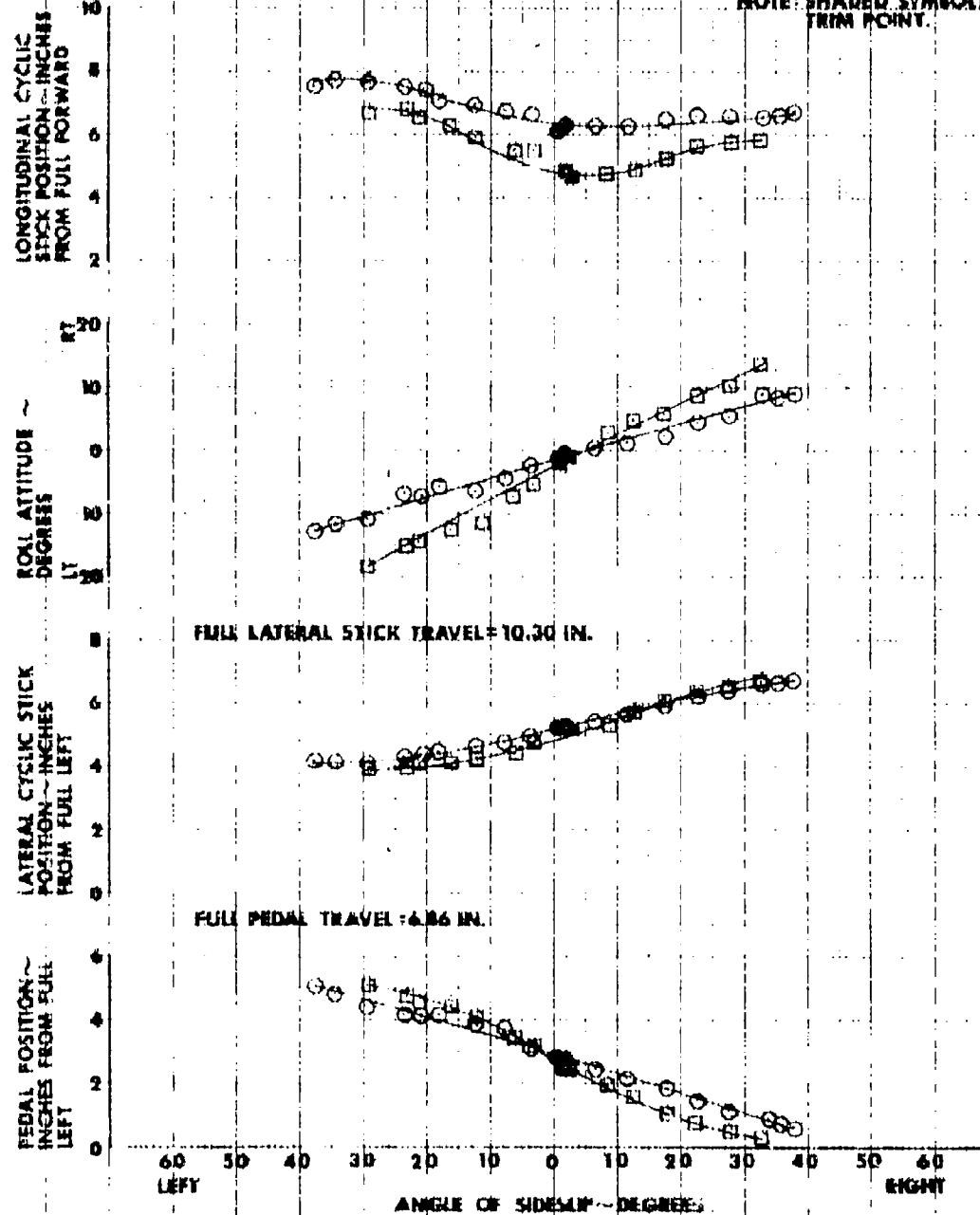


FIGURE 25  
STATIC LATERAL DIRECTIONAL STABILITY  
CH-53A USA S/N 68-10706

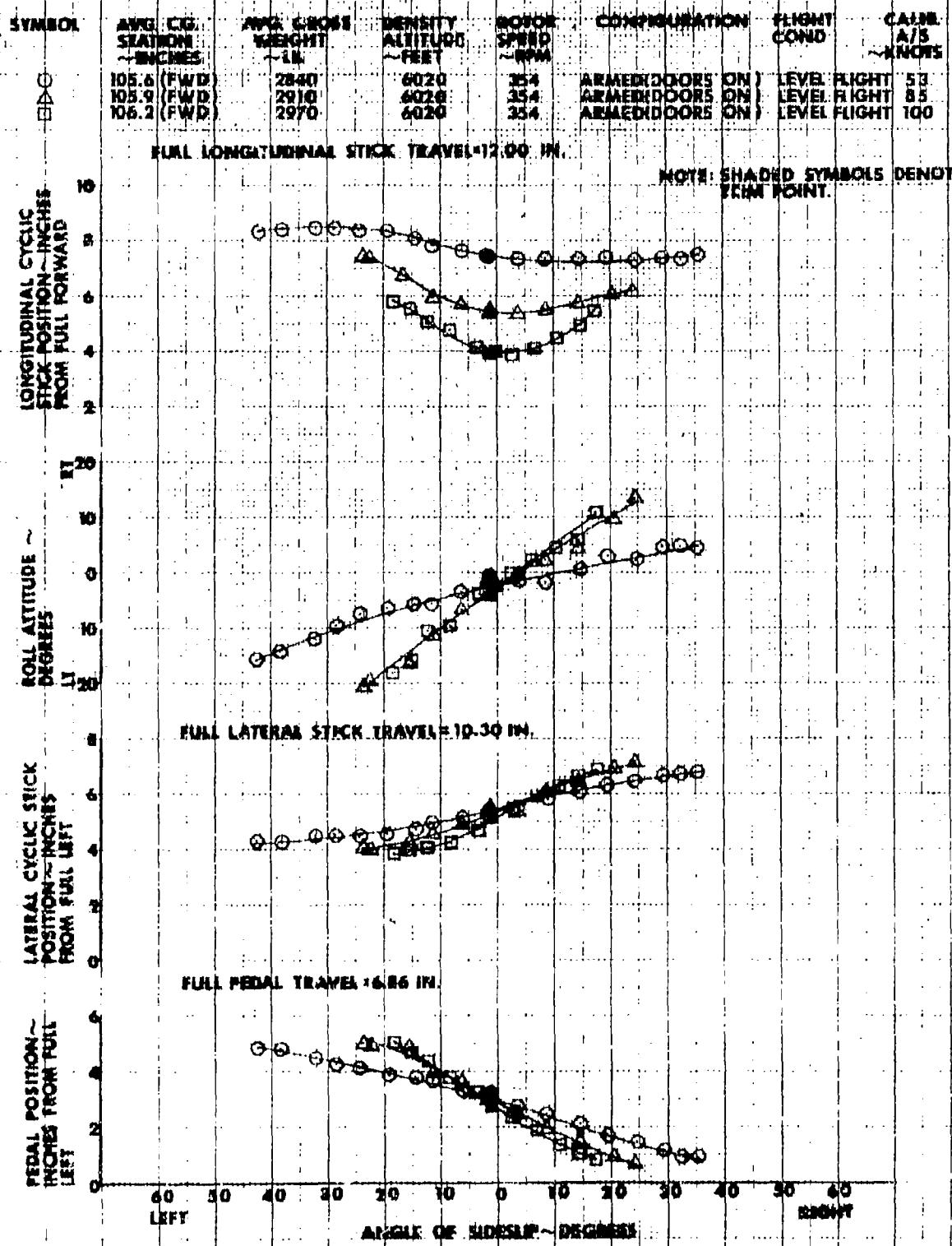


FIGURE 25  
STATIC LATERAL DIRECTIONAL STABILITY  
CH-53A USA 5/7/68 14700

SYMBOL	ANG. C.G. STATION INCHES	WING GROSS WEIGHT LBS.	DENSITY ALTITUDE FEET	ROTOR SPEED RPM	COMBINATION	WEIGHT COMBO	CALC. A/S IN KNOTS
112.1(AFT)	2960	5930	354	ARMED(DOORS ON)	LEVEL FLIGHT	53	
112.0(AFT)	2890	5930	354	ARMED(DOORS ON)	LEVEL FLIGHT	85	
112.0(AFT)	2890	5930	354	ARMED(DOORS ON)	LEVEL FLIGHT	103	

FULL LONGITUDINAL STICK TRAVEL=12.0 IN.

NOTE: SHADED SYMBOLS DENOTE  
TEAR POINT.

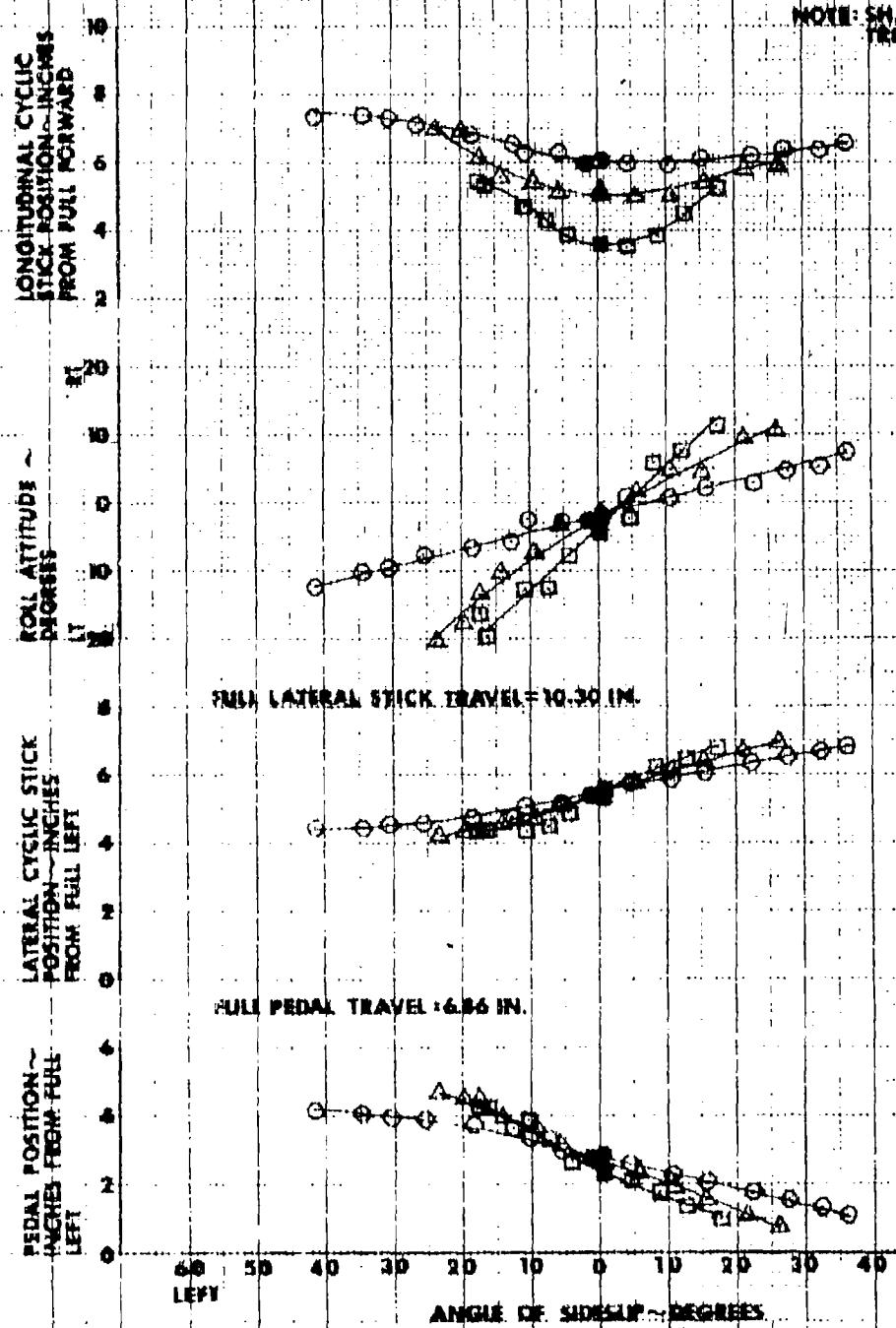


FIGURE 27  
STATIC LATERAL DIRECTIONAL STABILITY  
CH-33A USA S/N 68-10700

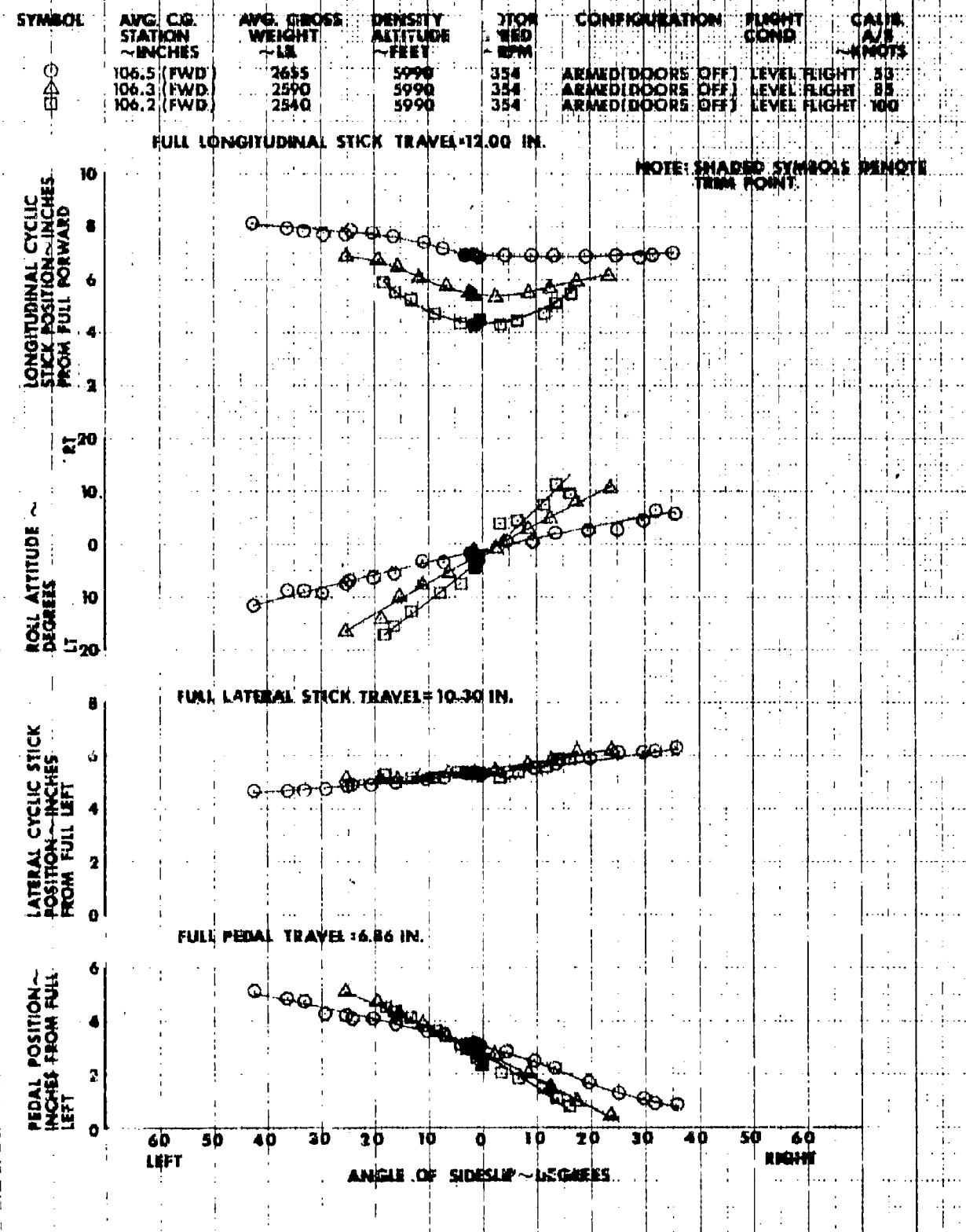


FIGURE 28  
STATIC LATERAL DIRECTIONAL STABILITY  
OH-58A USA S/N 68-16706

SYMBOL	Avg. C.G. STATION ~ INCHES	Avg. GROSS WEIGHT ~ LB.	DENSITY ~ PSEET	MOTOR SPEED ~ RPM	CONFIGURATION	FLIGHT COND	CALCS. A/5 ~ KNOTS
	105.7 (FWD)	2650	5970	354	CLEAN (DOORS ON)	LEVEL FLIGHT	83
	105.6 (FWD)	2600	5970	354	CLEAN (DOORS ON)	LEVEL FLIGHT	85
	105.4 (FWD)	2520	5970	354	CLEAN (DOORS ON)	LEVEL FLIGHT	113

FULL LONGITUDINAL STICK TRAVEL=17.00 IN.

NOTE: SHADED SYMBOLS DENOTE  
TRIM POINT.

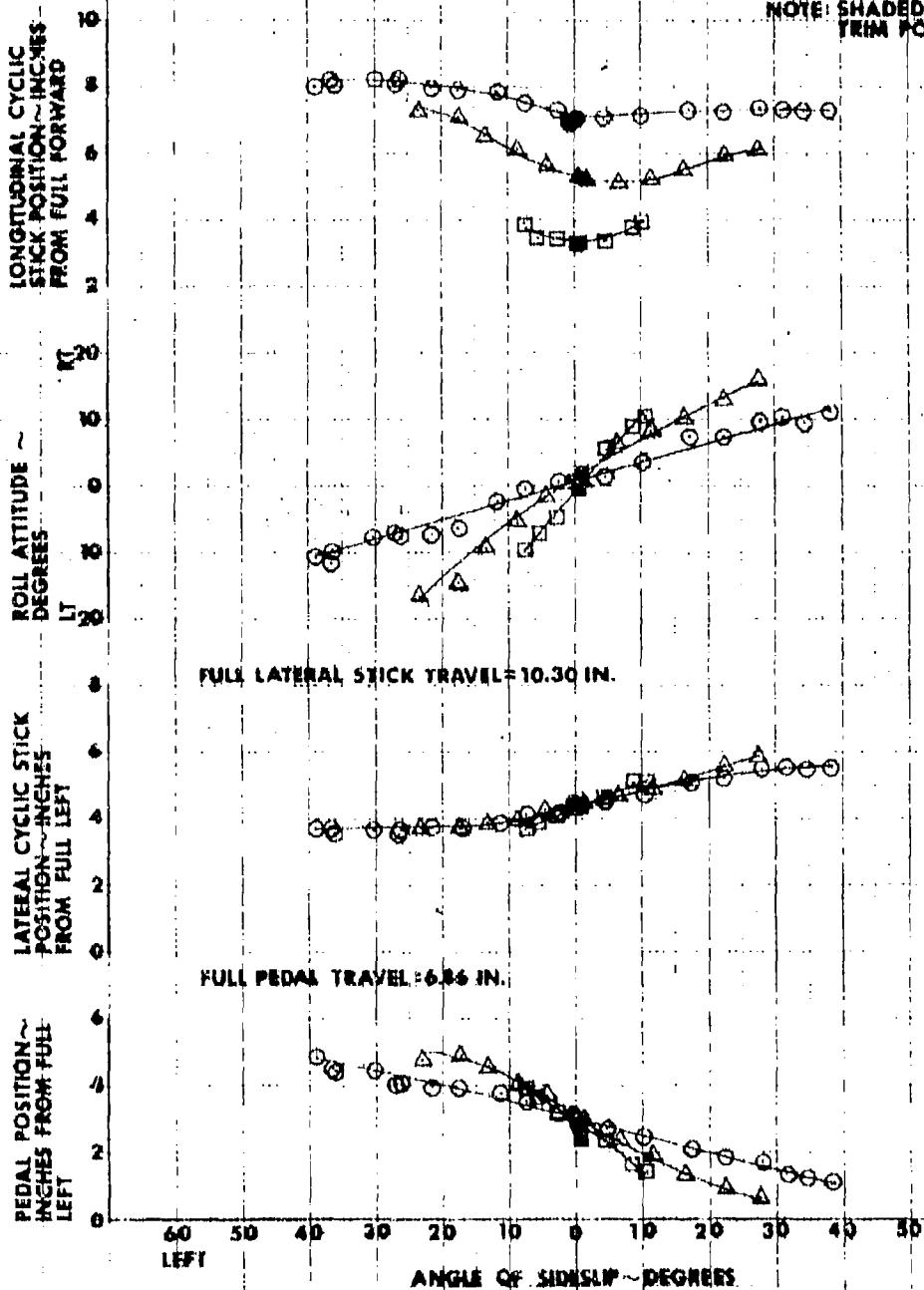
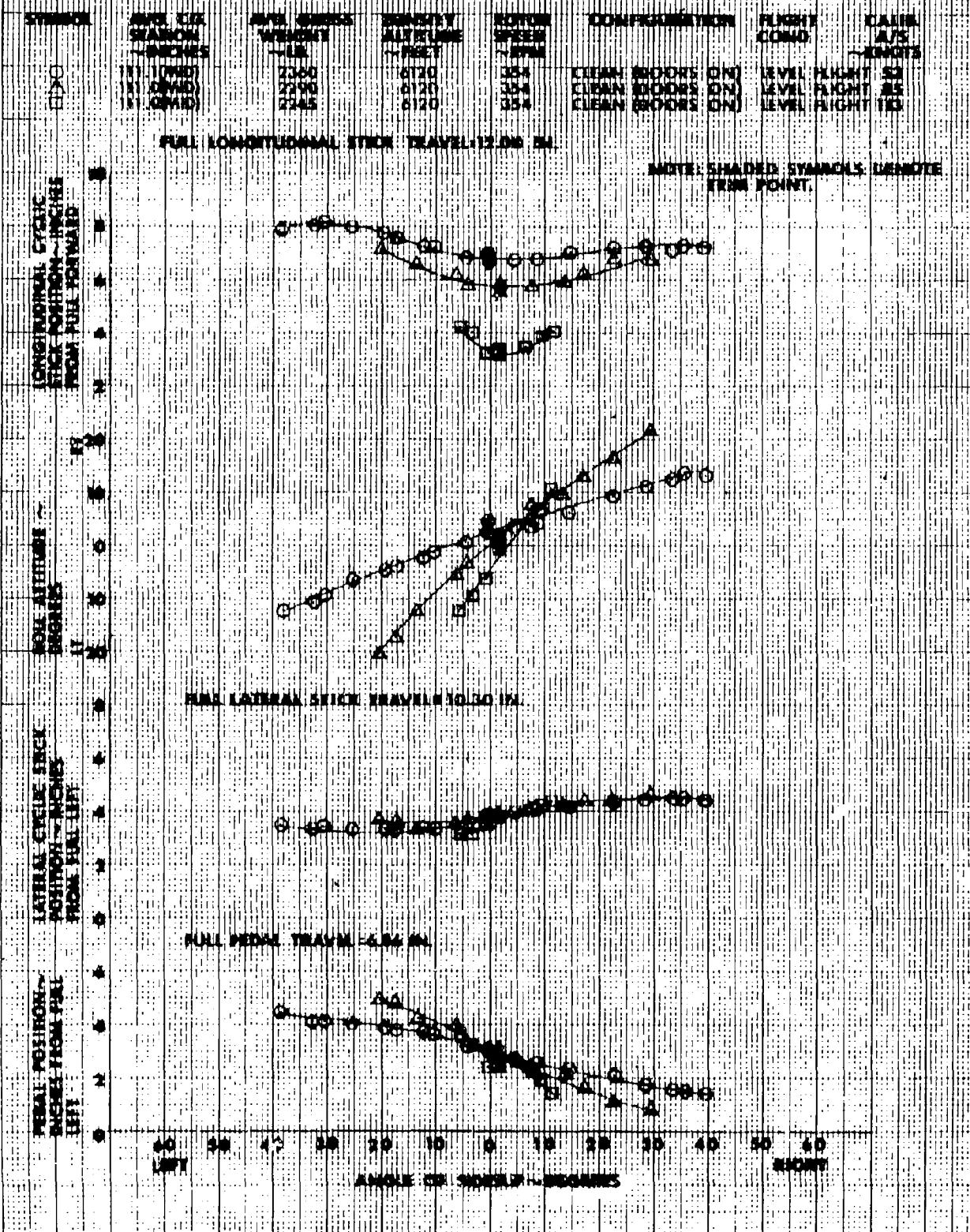
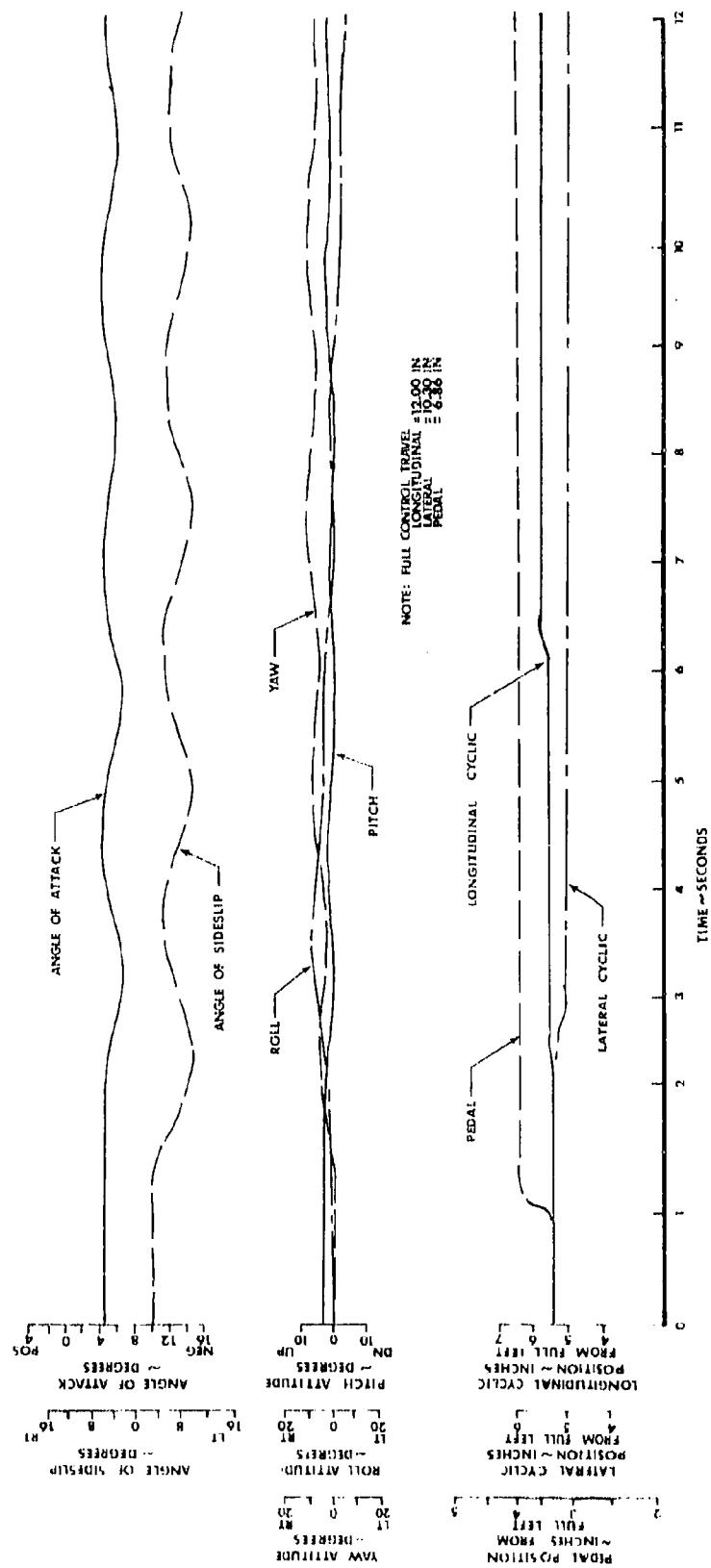


FIGURE 29  
STATIC LONGITUDINAL DIRECTIONAL STABILITY  
ON USA USA S/N 48-14704



**FIGURE 30**  
**DUTCH ROLL IN LEFT SIDESLIP**  
**OH-58A USA S/N 68-16706**  
**DENSITY ALTITUDE ~ 2950 FT**  
**CAULIBRATED AIRSPEED ~ 93 KTS**  
**MOTOR SPEED ~ 354 RPM**  
**GROSS WEIGHT ~ 2740 LB**  
**CG STATION ~ 105.3 IN (FWD)**  
**CONFIGURATION = ARMED (DOORS ON)**



**FIGURE 31**  
**LONGITUDINAL PULSE IN HIGH SPEED LEVEL FLIGHT**

OH-58A USA S/N 68-16706

LINISITY ALTITUDE ~ 5720 FT  
 CALIBRATED AIRSPEED ~ 354 KTS  
 ROTC R SPEED ~

GROSS WEIGHT ~ 2750 LB  
 CG STATION ~ 105.4 IN (FWD)  
 CONFIGURATION - CLEAN (DOORS ON)

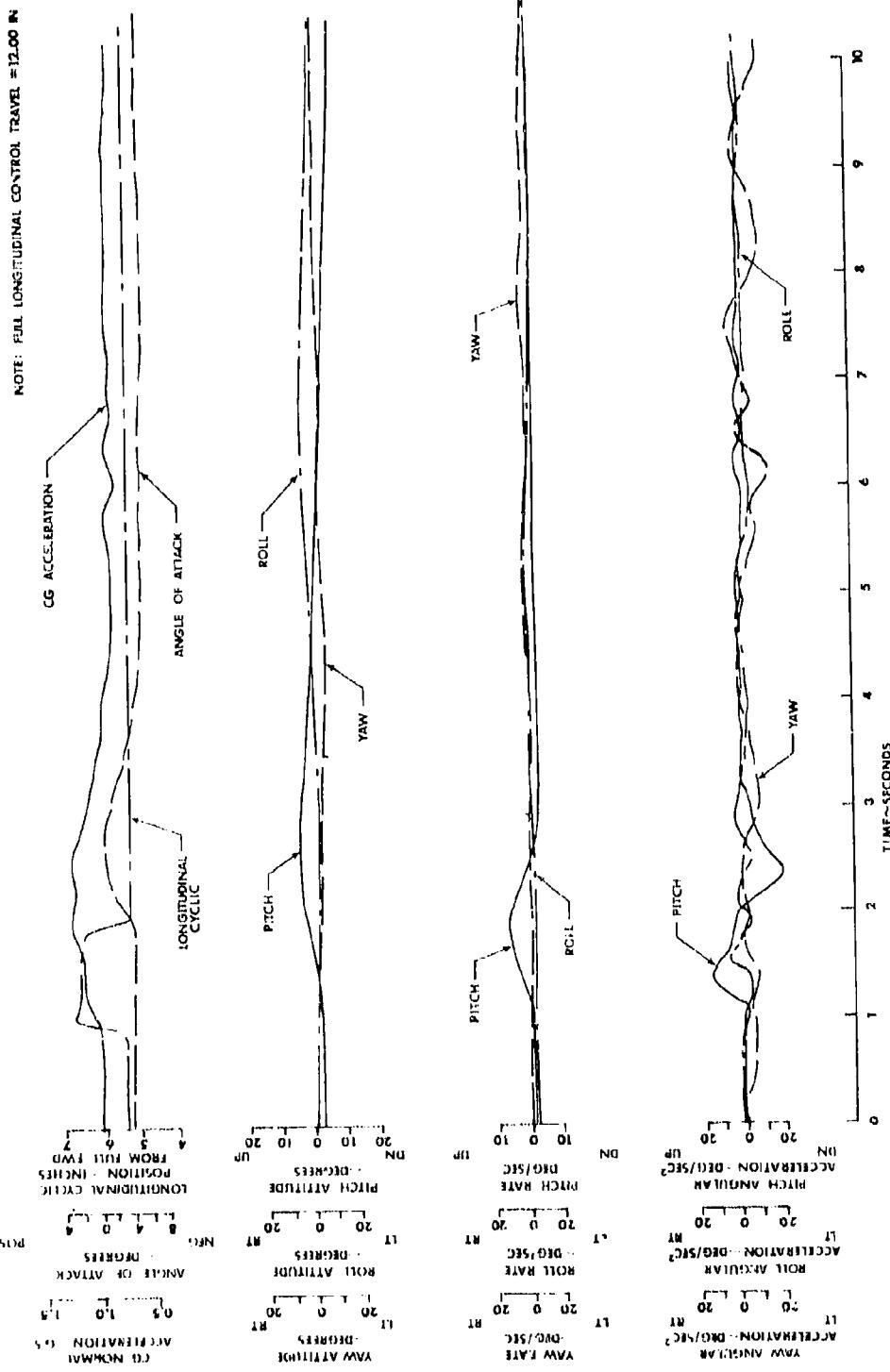
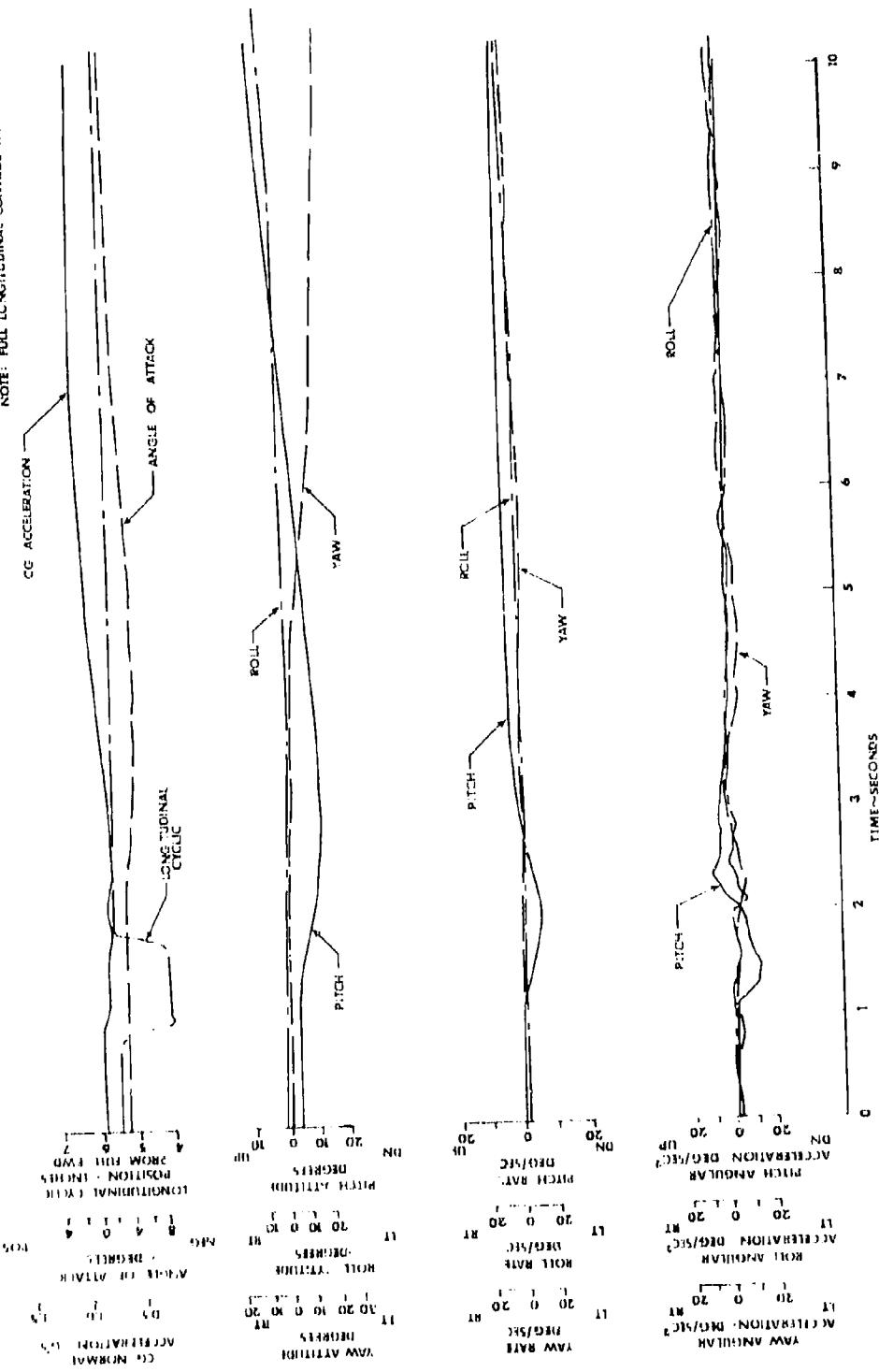


FIGURE 32  
LONGITUDINAL PULSE IN LOW SPEED LEVEL FLIGHT

OH-58A USA S/N 68-16706

GROSS WEIGHT ~ 2770 LB  
CG STATION ~ 105.6 IN (F+A)  
CONFIGURATION - CLEAN (DOORS ON)  
DENSITY ALTITUDE ~ 5120 FT  
CALIBRATED AIRSPEED ~ 38 KTS  
ROTOR SPEED ~ 354 RPM

NOTE: FULL LONGITUDINAL CONTROL TRAVEL = 12.00 IN



**FIGURE 33**  
**LATERAL PULSE IN HIGH SPEED LEVEL FLIGHT:**

OH-38A USA S/N 68-16706

DENSITY ALTITUDE = 5530 FT  
 CALIBRATED AIRSPEED = 104 KTS  
 ROTOR SPEED = 354 RPM

GROSS WEIGHT = 2450 LB  
 CG STATION = 705 IN (FWD)  
 CONFIGURATION = CLEAN (DOORS ON)

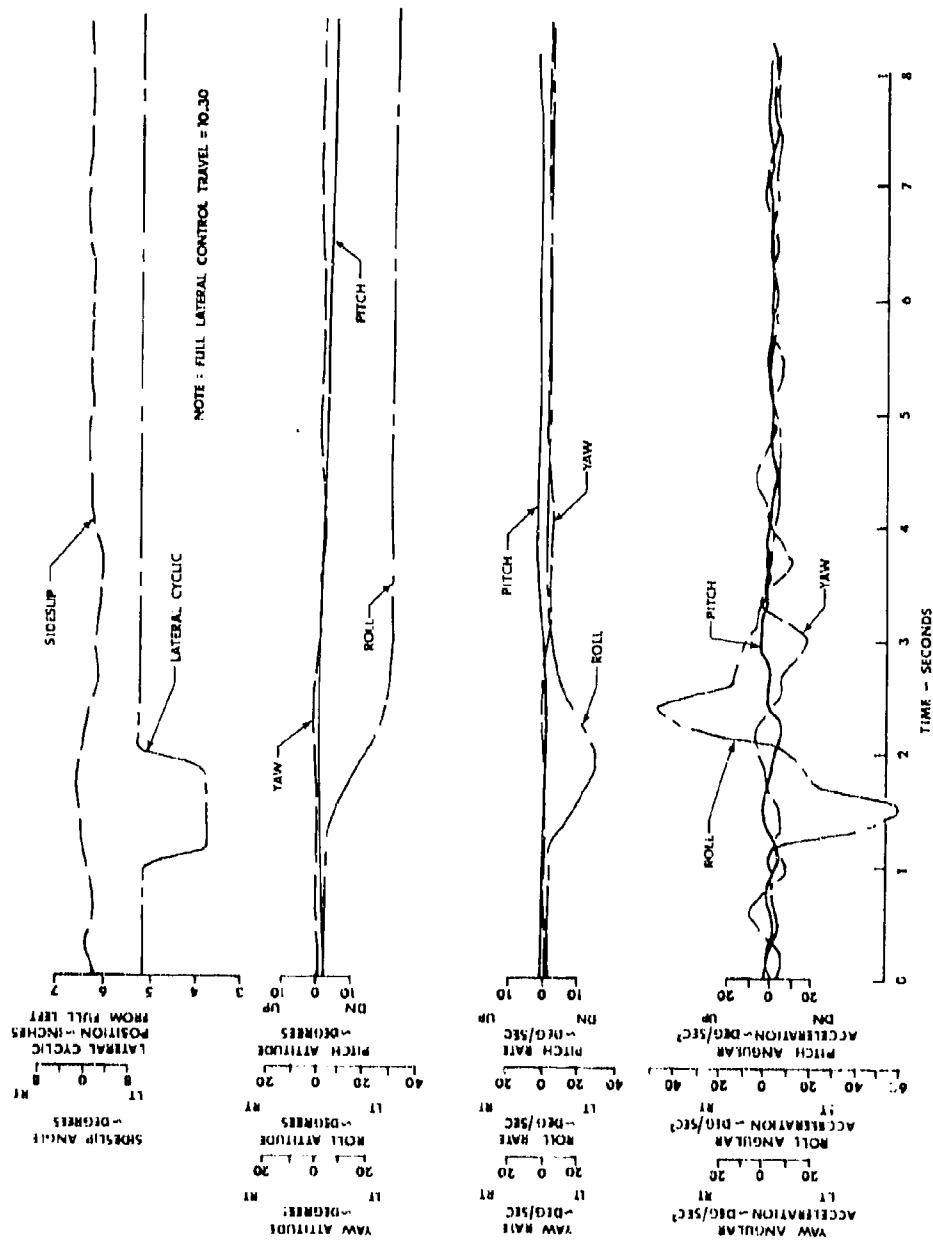
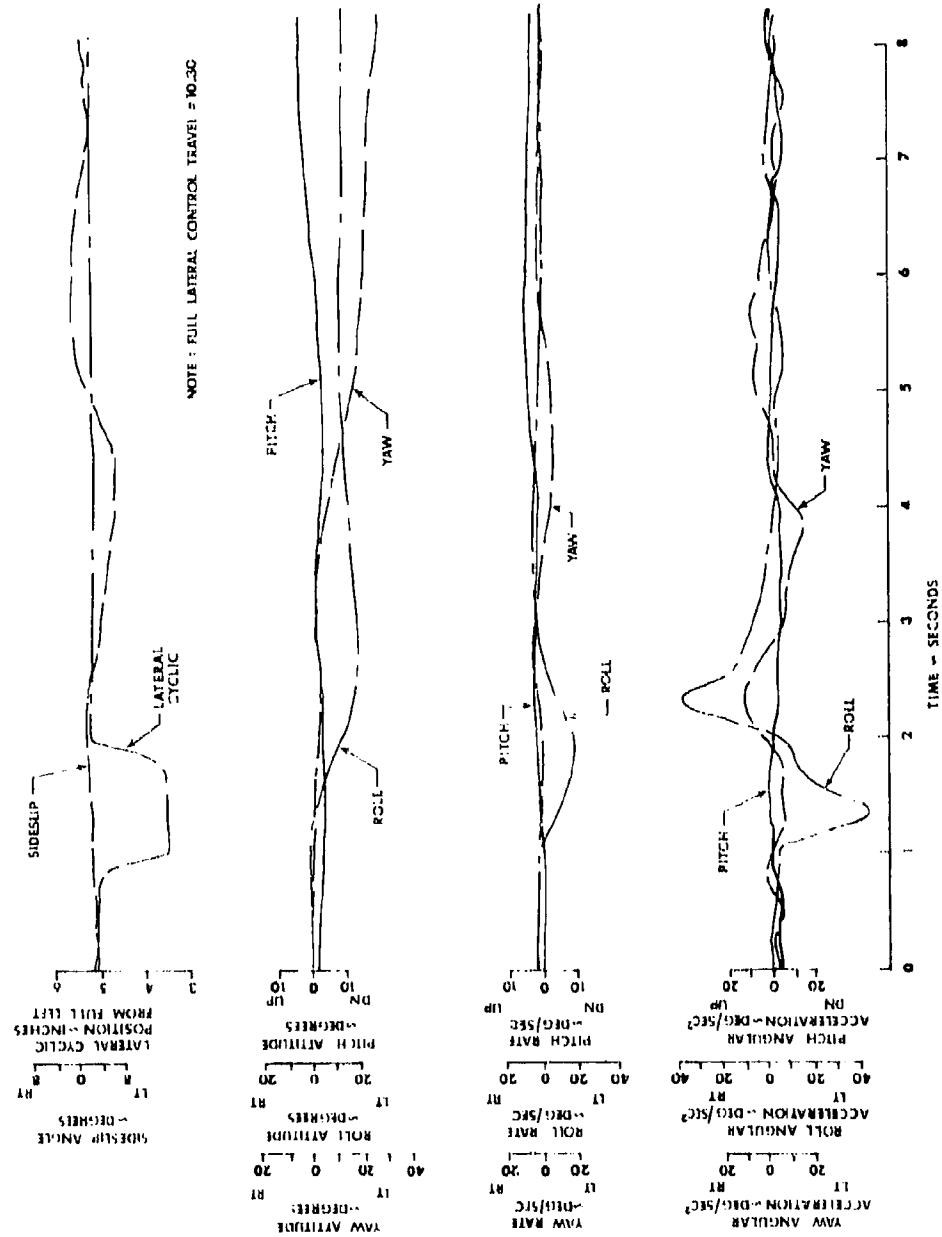


FIGURE 34  
LATERAL PULSE IN LOW SPEED LEVEL FLIGHT

OH-58A USA S/N 68-16706

DENSITY ALTITUDE = 5960 FT  
COLD AIR TEMP = 49 °F  
ROTOR SPEED = 354 RPM  
GROSS WEIGHT = 2770 LB  
CG POSITION = 107.0 IN FWD  
CONFIGURATION = ARMED (DOORS ON)



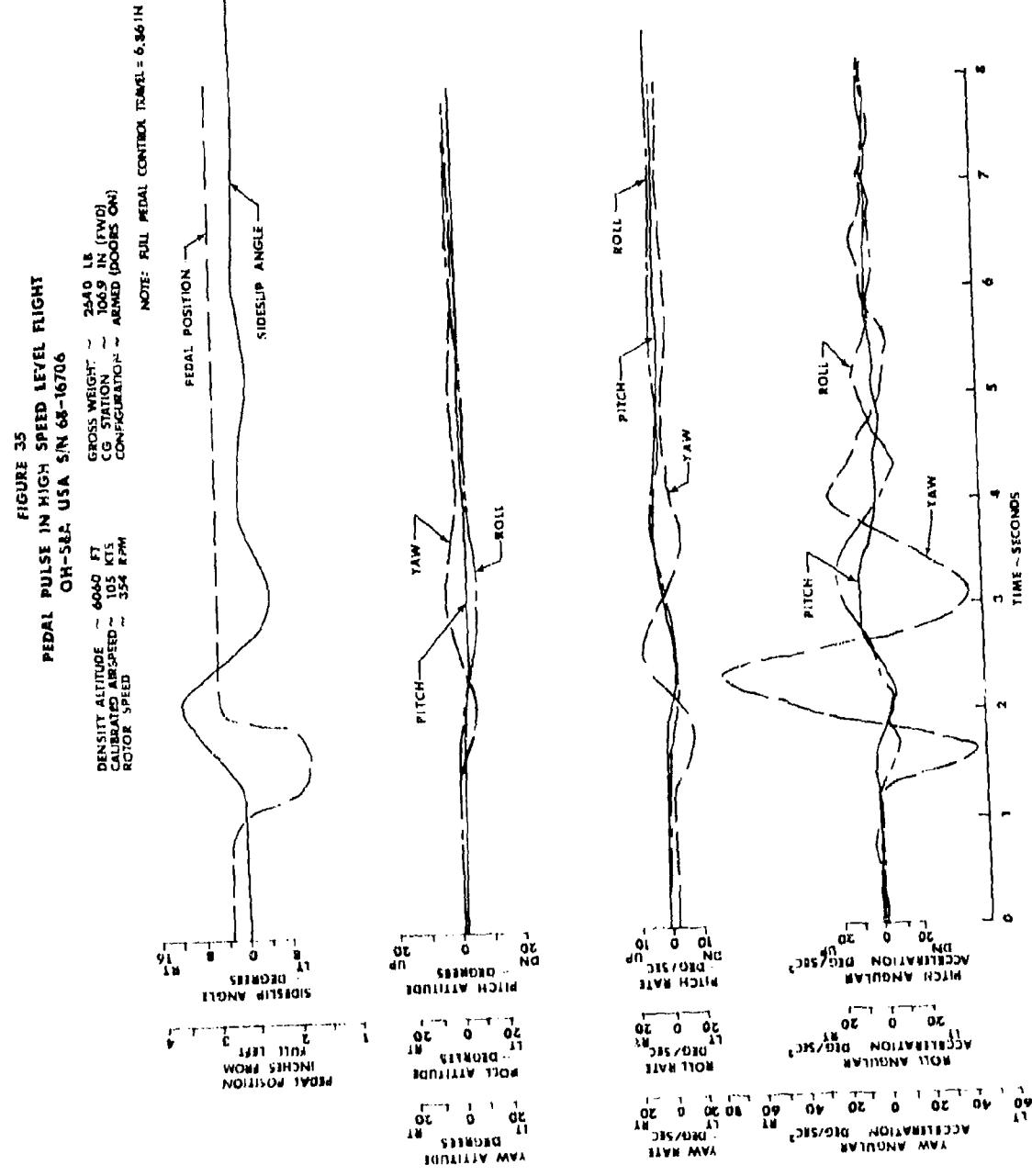
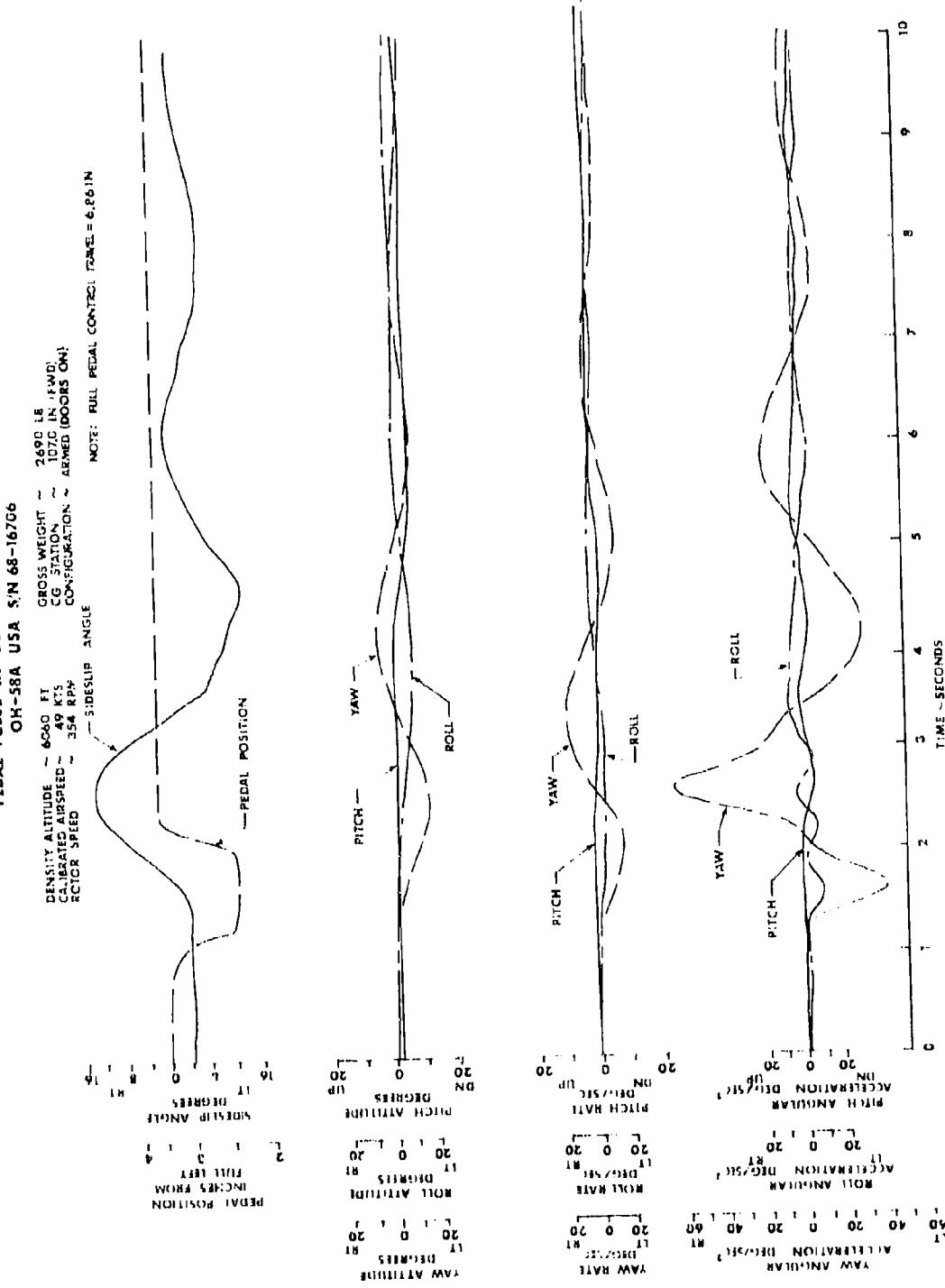


FIGURE 36  
FEDAI PULSE IN LOW SPEED LEVEL FLIGHT



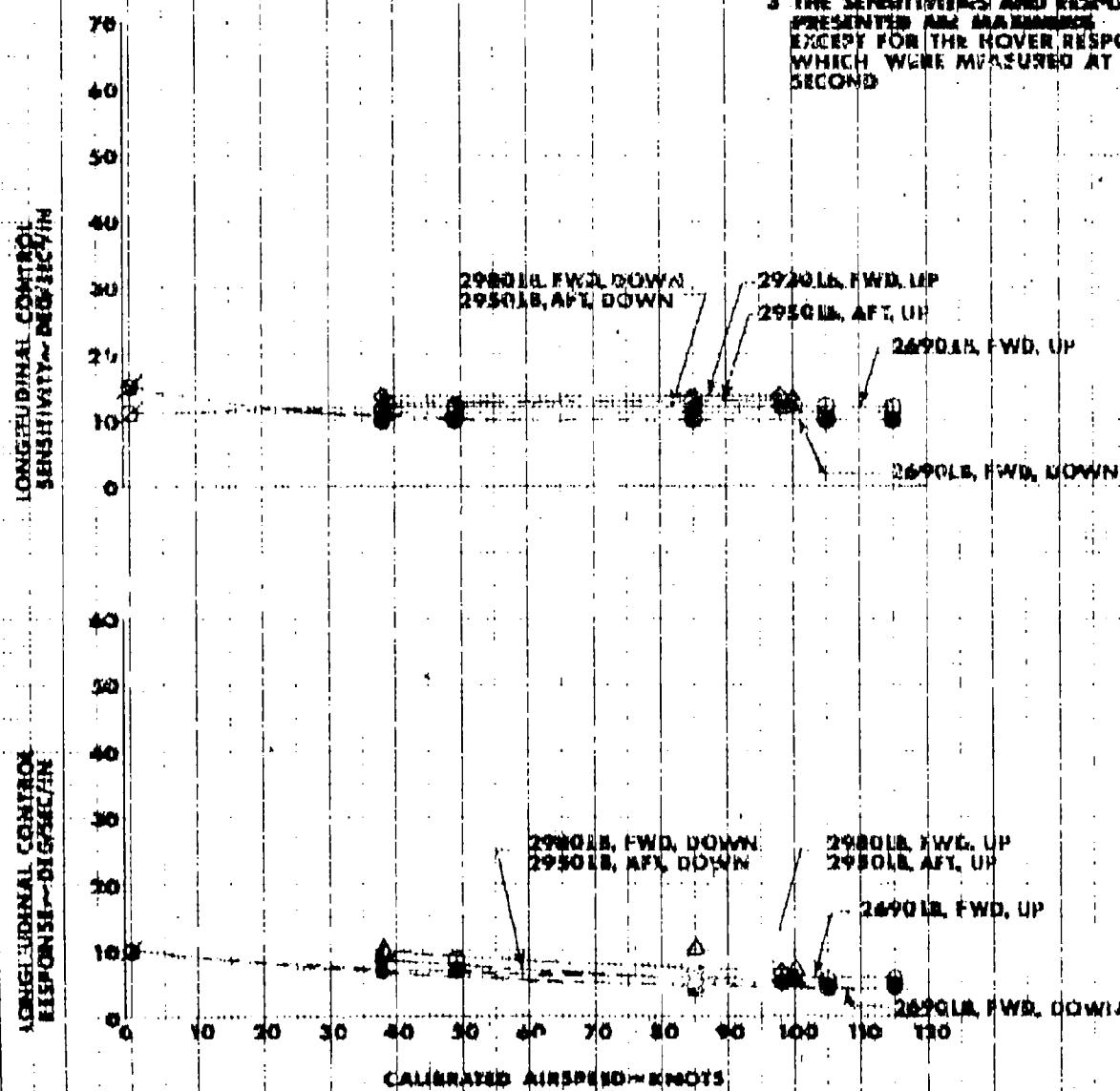
**FIGURE 37**  
**LONGITUDINAL SENSITIVITY AND RESPONSE SUMMARY**  
**OH-58A USA S/N 68-16706**

SYMBOL	Avg GROSS WEIGHT~LB	Avg CG STATION	DENSITY ~FT	ROTOR SPEED ~RPM	FLIGHT CONDITION
○	2690	107.0 FWD	5960	354	LEVEL FLIGHT
△	2645	106.6 FWD	3910	354	HOVER
□	2950	112.1 AFT	6140	354	LEVEL FLIGHT
◆	2980	106.3 FWD	6320	354	LEVEL FLIGHT

GROSS WEIGHT AND CENTER  
OF GRAVITY COMPARISON

NOTE:

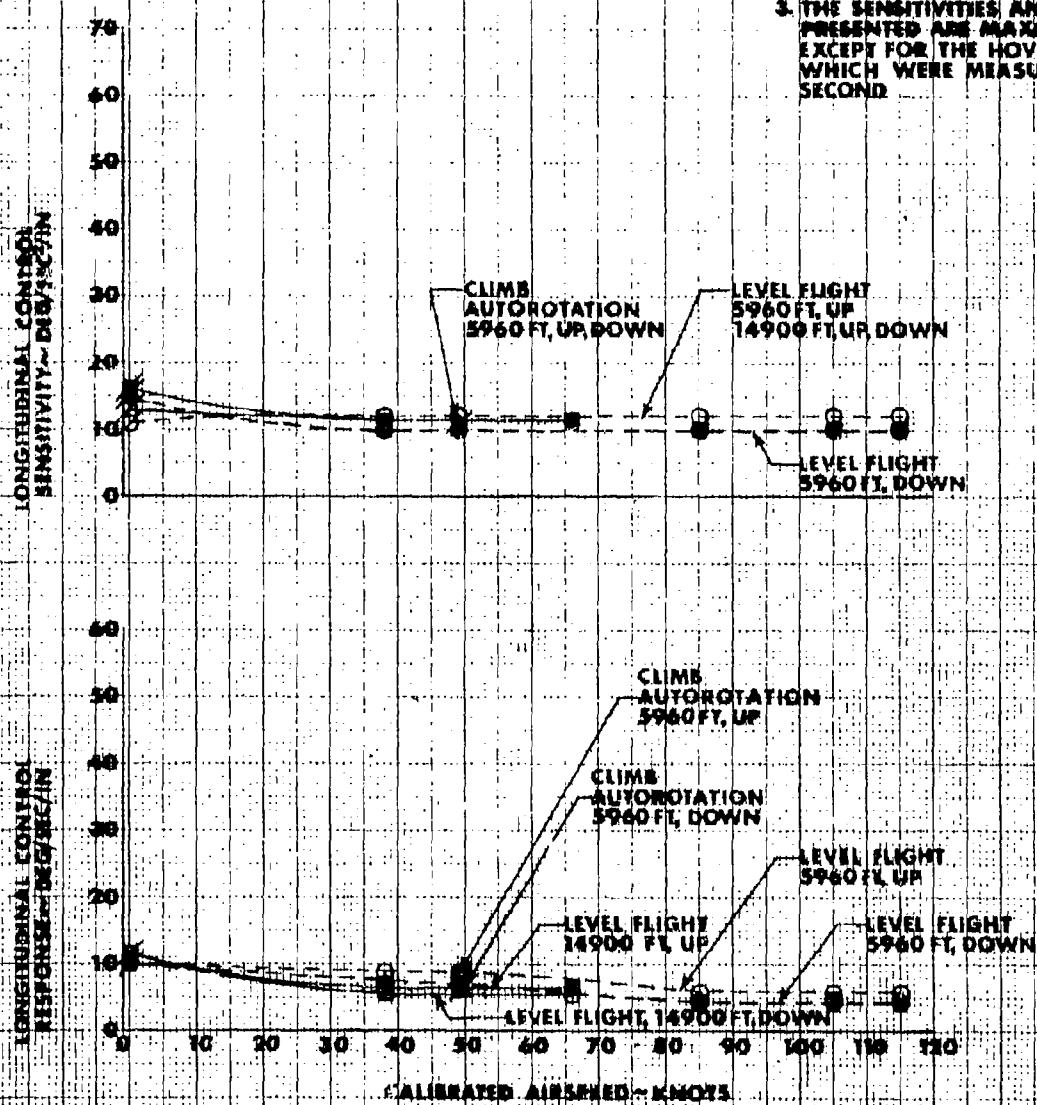
1. ARMED DOORS ON  
CONFIGURATION
2. OPEN SYMBOLS DENOTE  
PITCH UP  
SHADeD SYMBOLS DENOTE  
PITCH DOWN
3. THE SENSITIVITIES AND RESPONSES  
PRESENTED ARE MAXIMUM  
EXCEPT FOR THE HOVER RESPONSES  
WHICH WERE MEASURED AT 1  
SECOND



**FIGURE 38.  
LONGITUDINAL SENSITIVITY AND RESPONSE SUMMARY  
CH-53A USA S/N 68-10704**

SYMBOL	Avg GROSS WEIGHT~LB	Avg CG STATION	DENSITY ALTITUDE ~FT	ROTIN SPEED ~RPM	FLIGHT CONDITION
O	2690	107.0 FWD	5960	354	LEVEL FLIGHT
○	2662	106.8 FWD	3910	354	HOVER
□	2700	107.0 FWD	14900	354	LEVEL FLIGHT
■	2535	109.8 MID	10830	354	HOVER
△	2600	106.8 FWD	6080	354	CLIMB
△	2600	106.8 FWD	6080	354	AUTOROTATION

**ALTITUDE AND FLIGHT  
CONDITION COMPARISON**



NOTE:

1. ARMED DOORS ON CONFIGURATION
2. OPEN SYMBOLS DENOTE PITCH UP  
SMALL SYMBOLS DENOTE PITCH DOWN
3. THE SENSITIVITIES AND RESPONSES PRESENTED ARE MAXIMUM EXCEPT FOR THE HOVER RESPONSES WHICH WERE MEASURED AT 1 SECOND

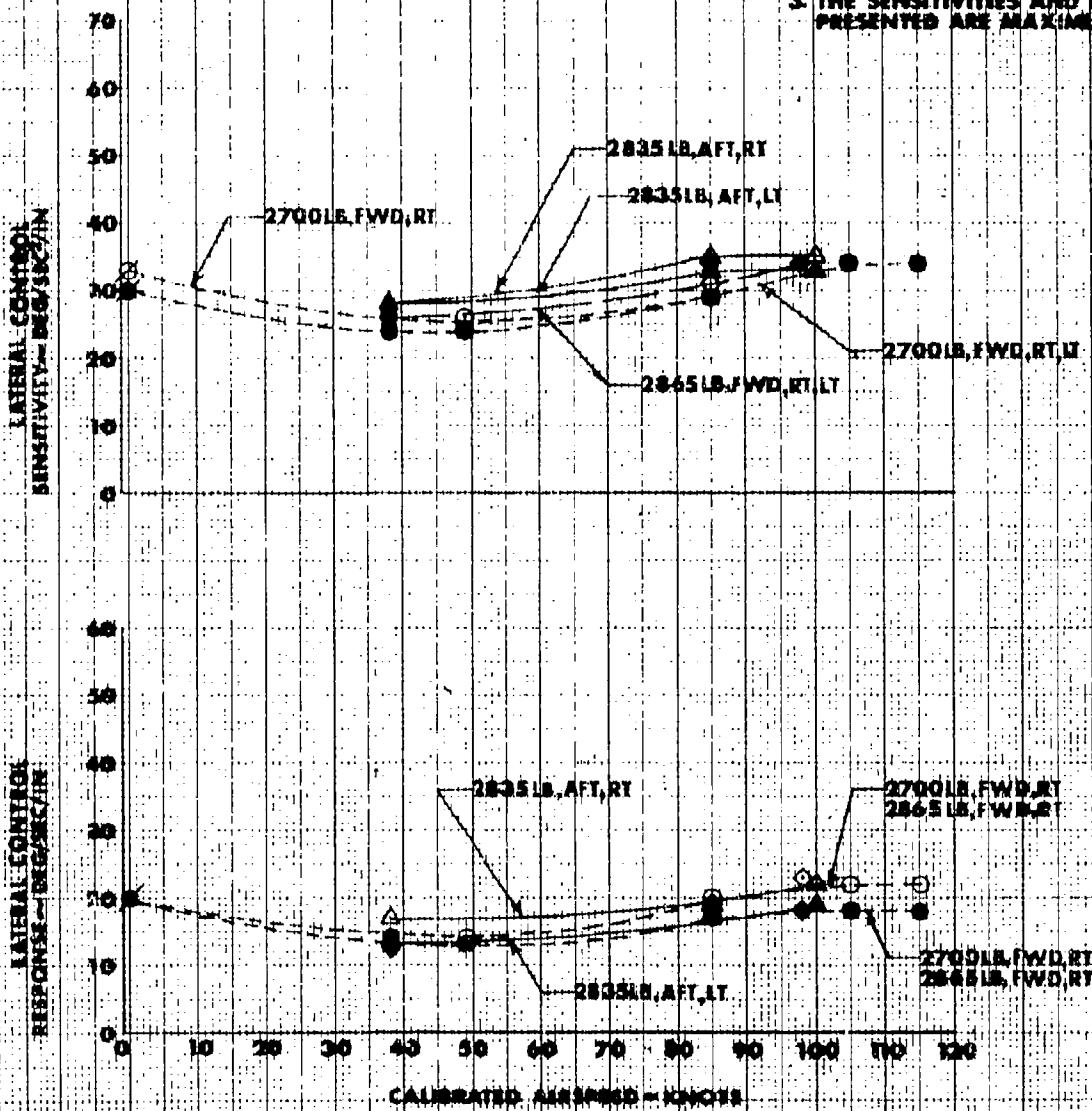
**FIGURE 39**  
**LATERAL SENSITIVITY AND RESPONSE SUMMARY**  
**OH-58A USA S/N 68-16706**

SYMBOL	Avg GROSS WEIGHT - LB	Avg CG STATION - IN	DENSITY - FT	MOTOR SPEED - RPM	FLIGHT CONDITION
○	2700	107.0	5960	354	LEVEL FLIGHT
△	2685	106.2	3910	354	HOVER
▲	2835	112.0	6140	354	LEVEL FLIGHT
◊	2865	105.6	6320	354	LEVEL FLIGHT

GROSS WEIGHT AND CENTER OF GRAVITY COMPARISON

NOTE:

1. ARMED DOORS ON CONFIGURATION
2. OPEN SYMBOLS DENOTE RIGHT ROLL
3. SHADDED SYMBOLS DENOTE LEFT ROLL
3. THE SENSITIVITIES AND RESPONSES PRESENTED ARE MAXIMUMS

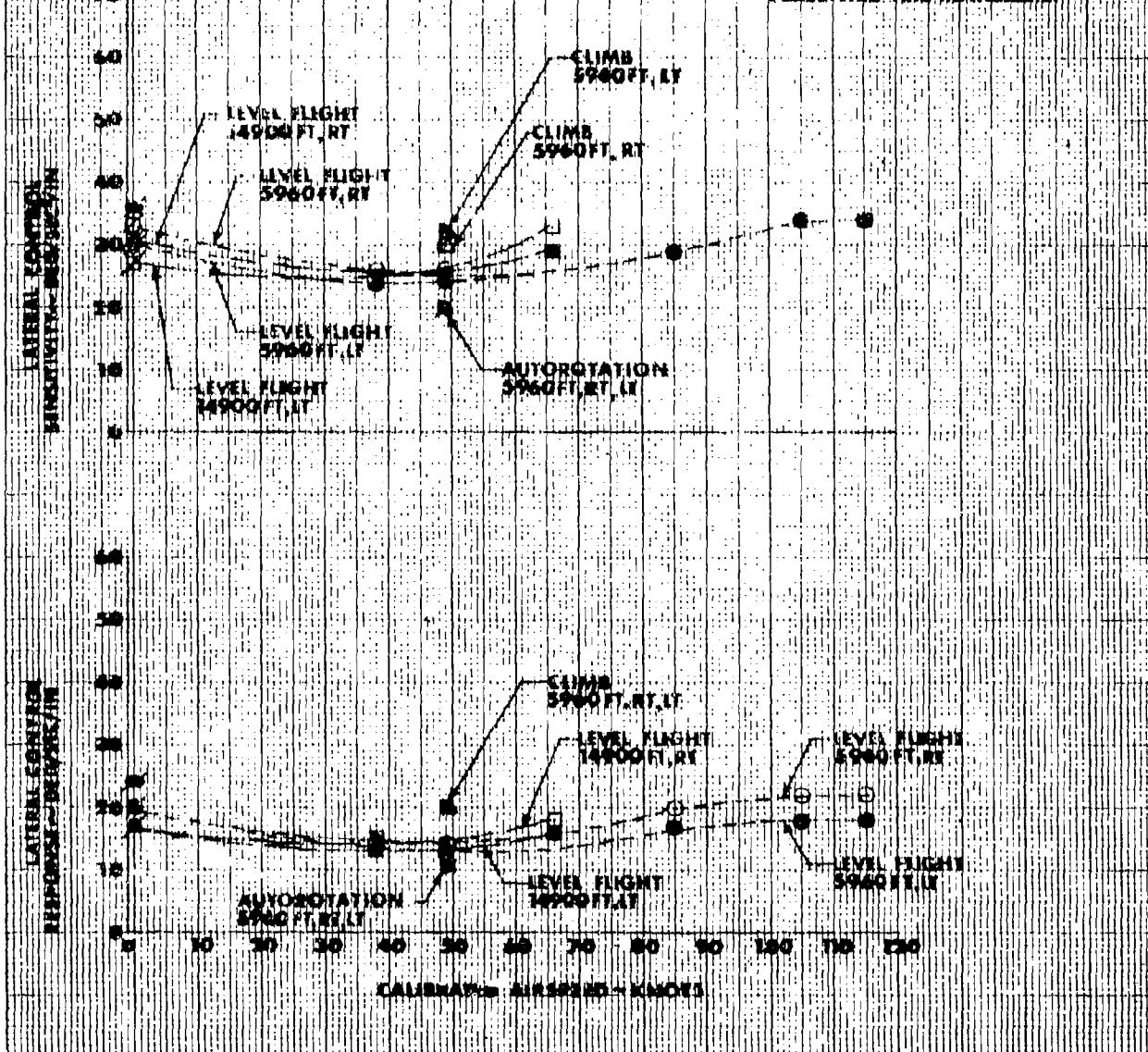


**FIGURE 10**  
**LATERAL SENSITIVITY AND RESPONSE SUMMARY**  
**OM-32A UO 5/W SP-19704**

SYMBOL	Avg GROSS WEIGHT-LB	Avg CP STATION	ALTITUDE FT	MOTOR SPEED	FLIGHT CONDITION
O	2700	107.0 FWD	5960	384	LEVEL FLIGHT
O	2485	106.7 FWD	2910	384	HOVER
O	2460	106.6 FWD	14200	384	LEVEL FLIGHT
J	2510	109.1 MID	108.30	384	HOVER
J	2485	106.1 FWD	-30	384	HOVER
J	2440	106.8 FWD	2960	384	CLIMB
J	2440	106.6 FWD	5960	384	AUTOROTATION

NOTE:

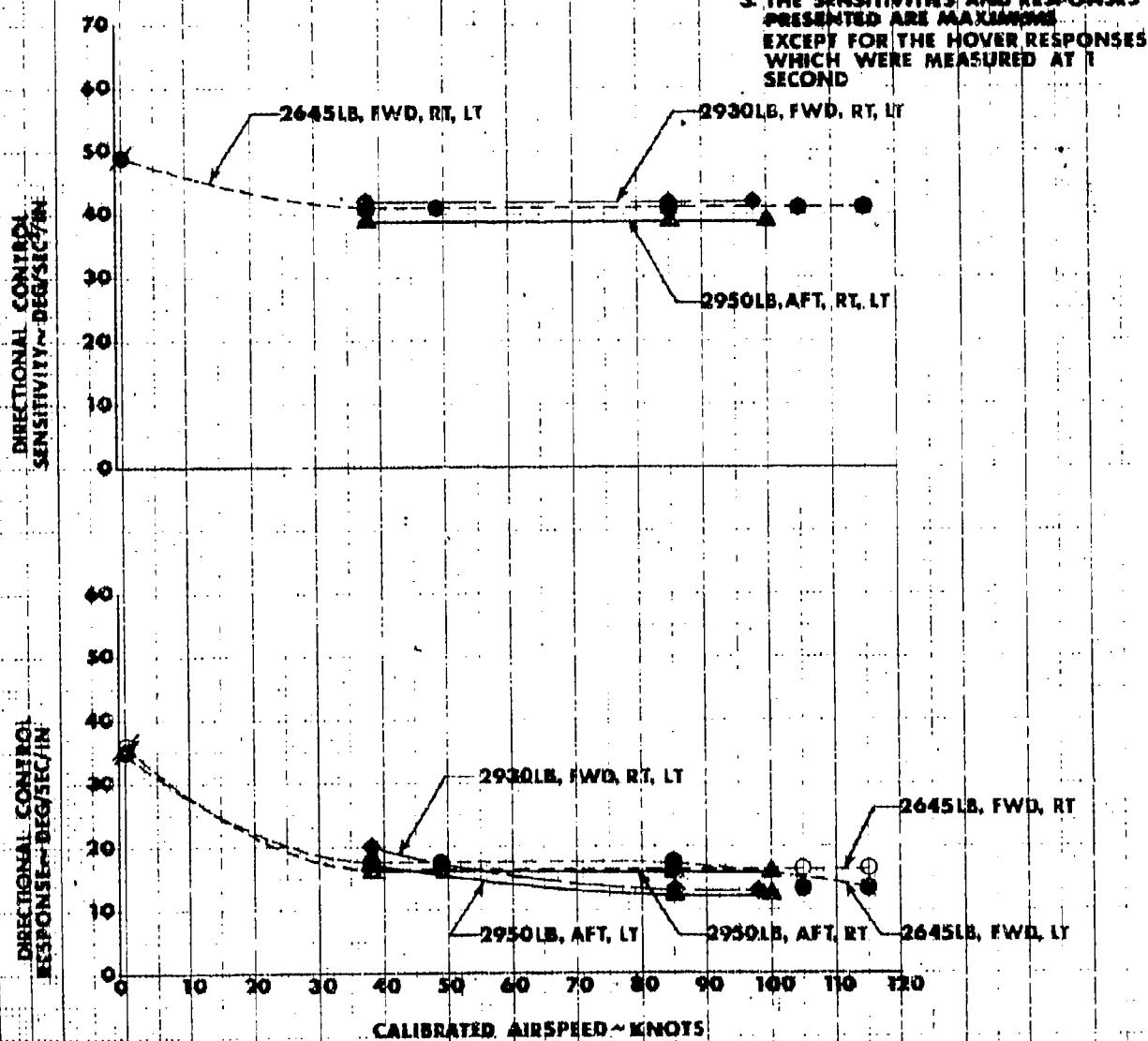
1. ARMED MOTORS ON CONFIGURATION
2. OPEN SYMBOLS DENOTE RIGHT ROLL
3. SHADeD SYMBOLS DENOTE LEFT ROLL
4. THE SENSITIVITIES AND RESPONSES PRESENTED ARE MAXIMUM



**FIGURE 41**  
**DIRECTIONAL SENSITIVITY AND RESPONSE SUMMARY**  
**OH-58A USA S/N 68-16706**

SYMBOL	Avg GROSS WEIGHT~LB	Avg CG STATION ~IN	DENSITY STATION	ROTOR ALTITUDE ~FT	SPEED ~RPM	FLIGHT CONDITION
○	2645	106.9	FWD	6060	354	LEVEL FLIGHT
○	2640	106.2	FWD	4450	354	HOVER
△	2950	112.1	AFT	5780	354	LEVEL FLIGHT
△	2930	106.0	FWD	6130	354	LEVEL FLIGHT

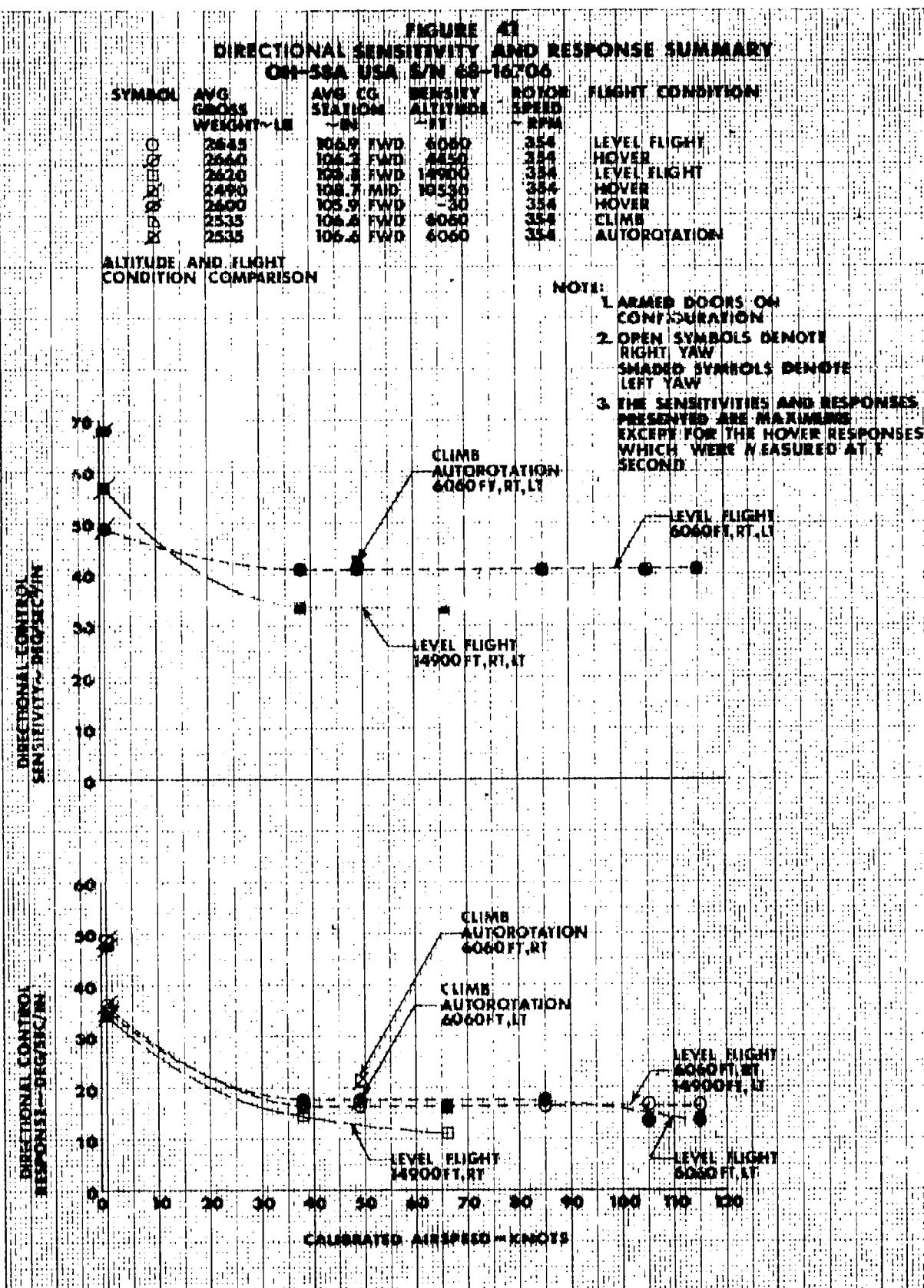
GROSS WEIGHT AND CENTER OF GRAVITY COMPARISON

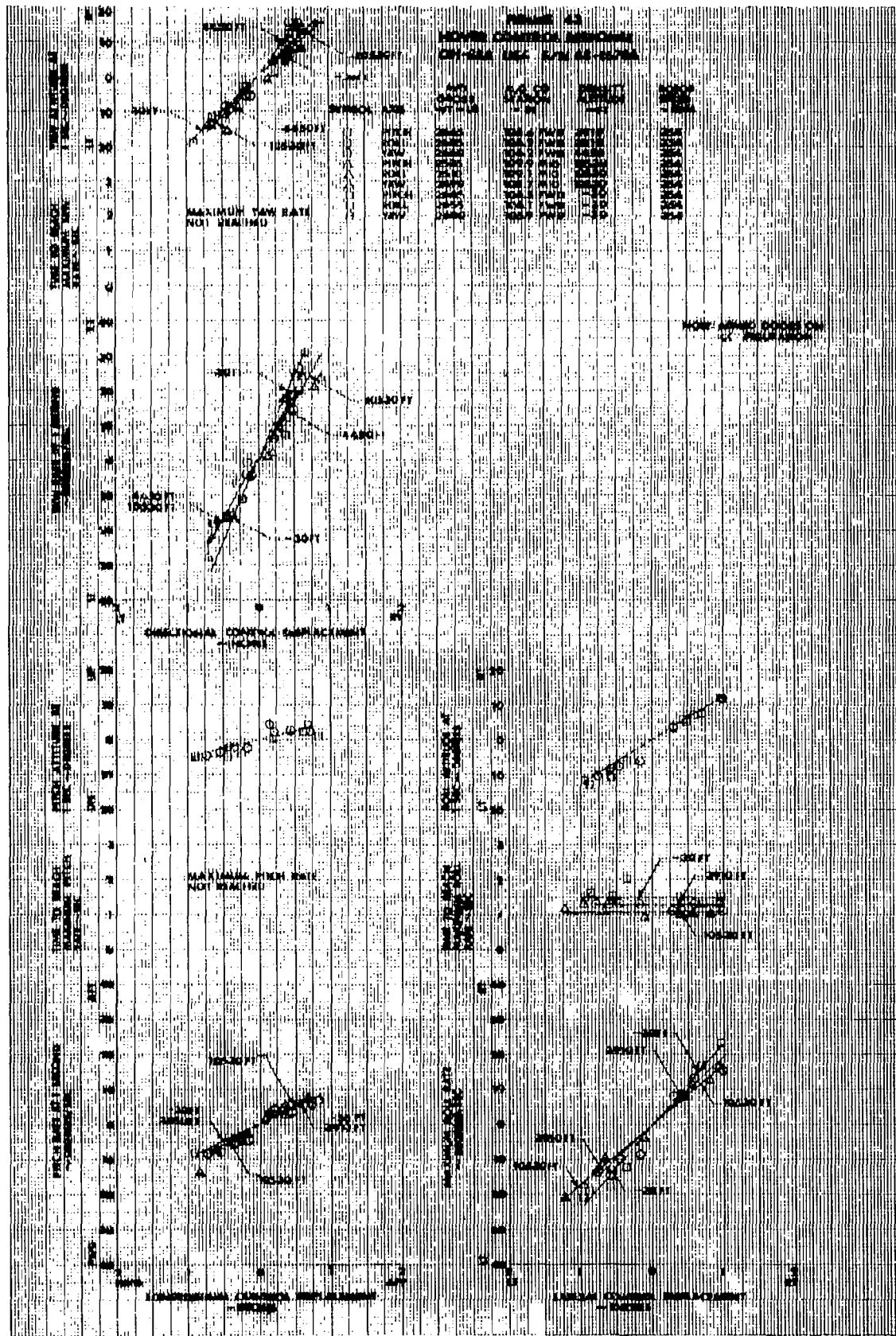


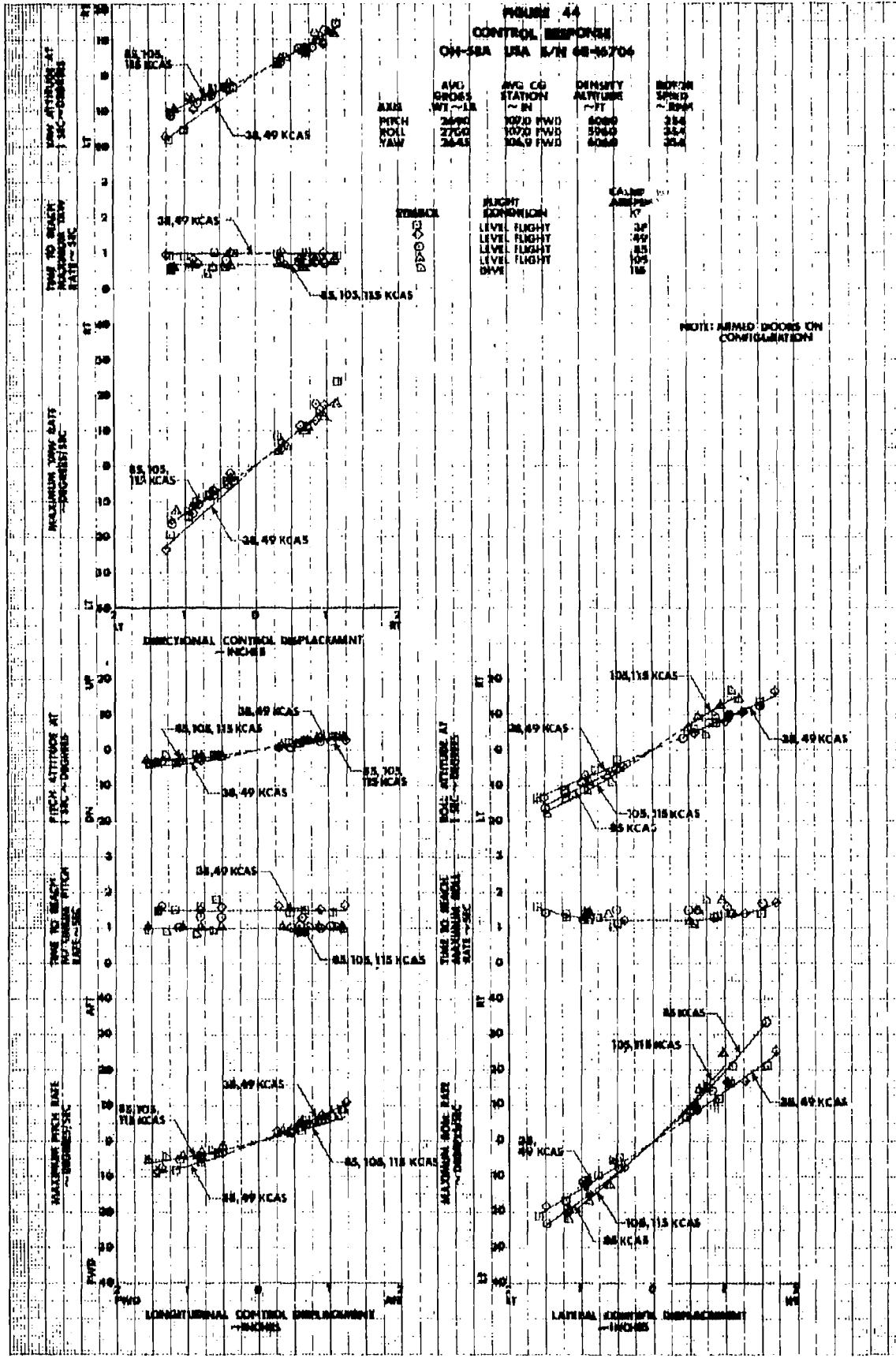
NOTE:

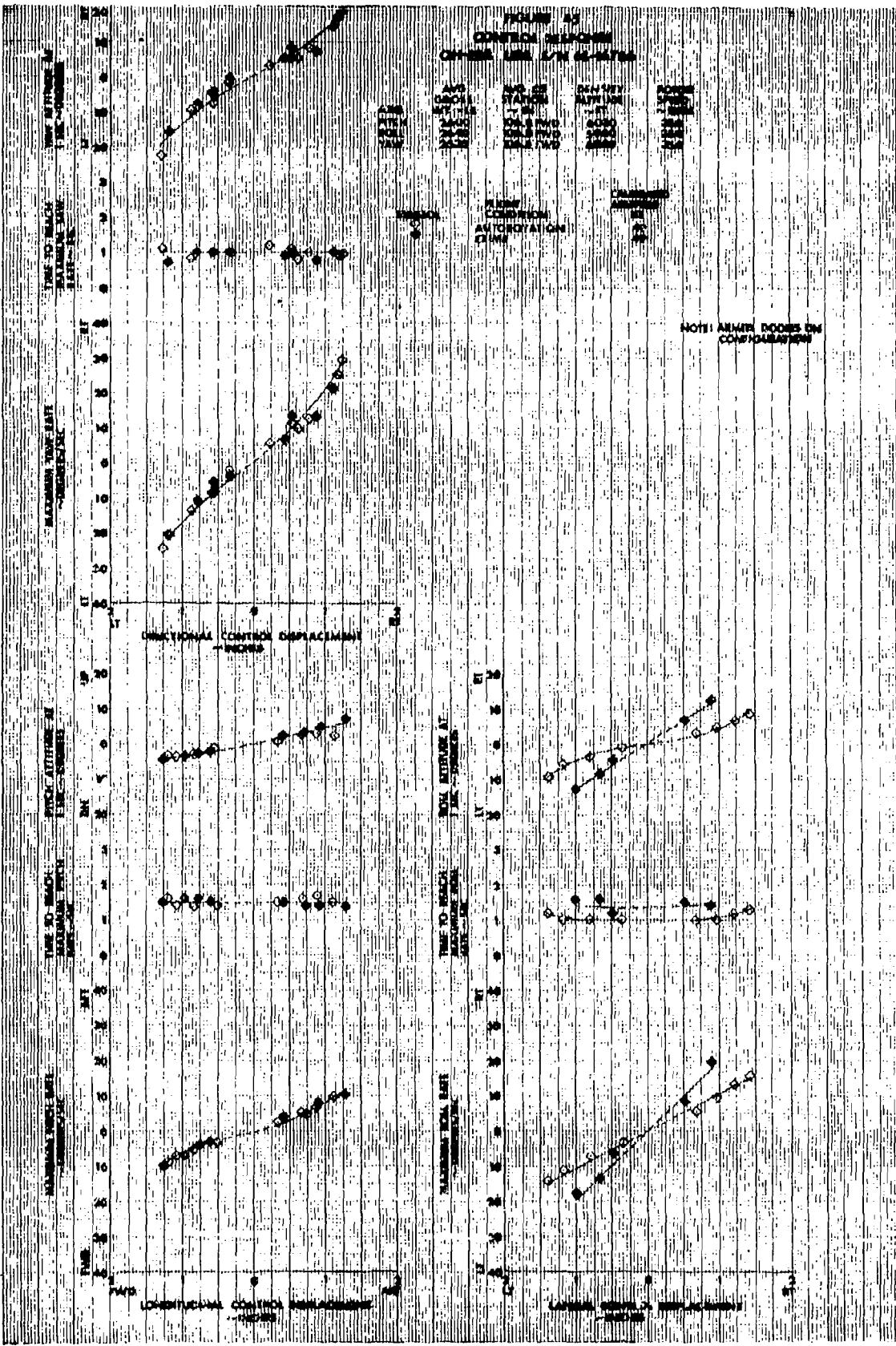
1. ARMED DOORS ON CONFIGURATION
2. OPEN SYMBOLS DENOTE RIGHT YAW  
SHADDED SYMBOLS DENOTE LEFT YAW
3. THE SENSITIVITIES AND RESPONSES PRESENTED ARE MAXIMUM EXCEPT FOR THE HOVER RESPONSES WHICH WERE MEASURED AT 1 SECOND

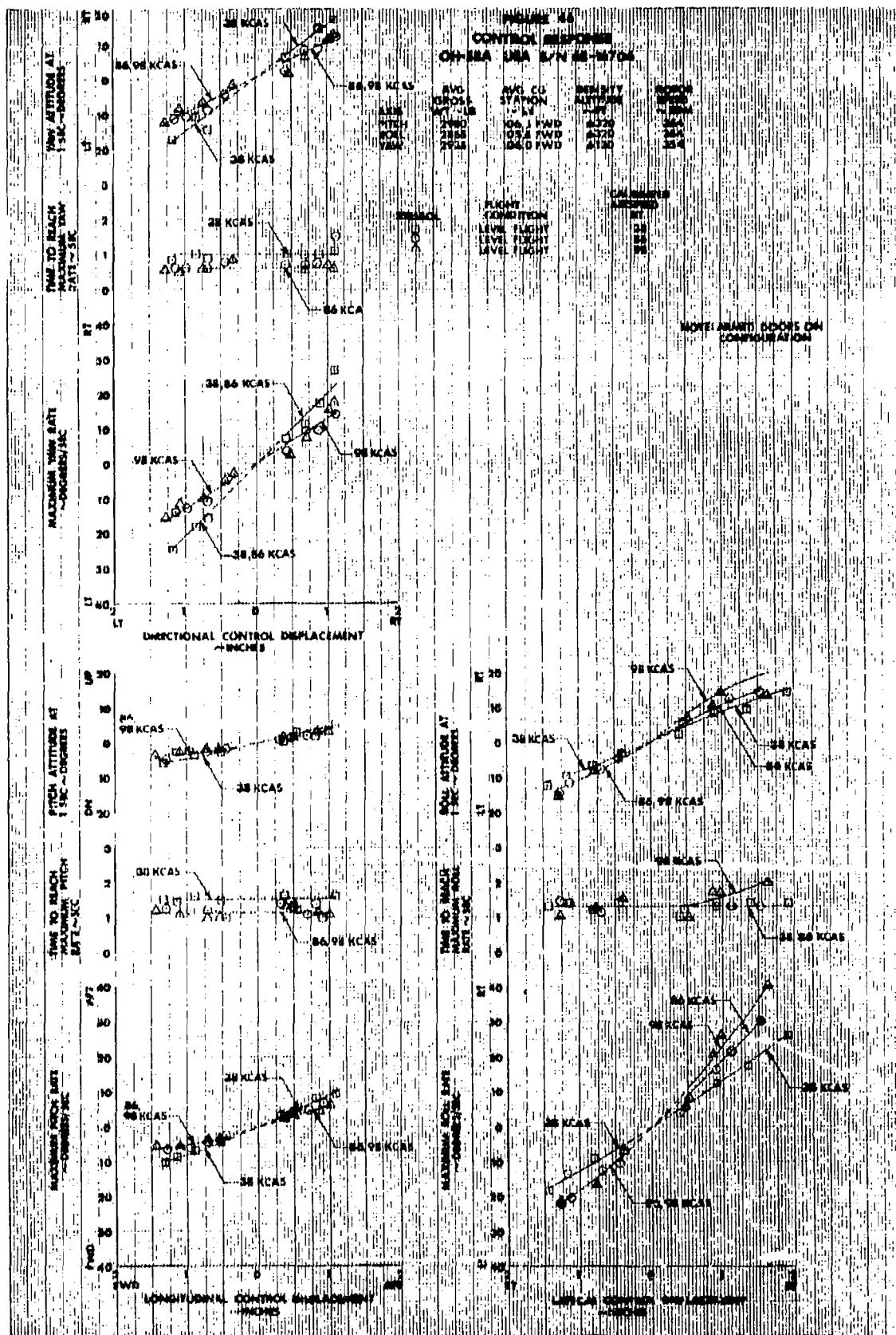
**FIGURE 43**  
**DIRECTIONAL SENSITIVITY AND RESPONSE SUMMARY**  
**OH-58A USA 6/14/69 16704**











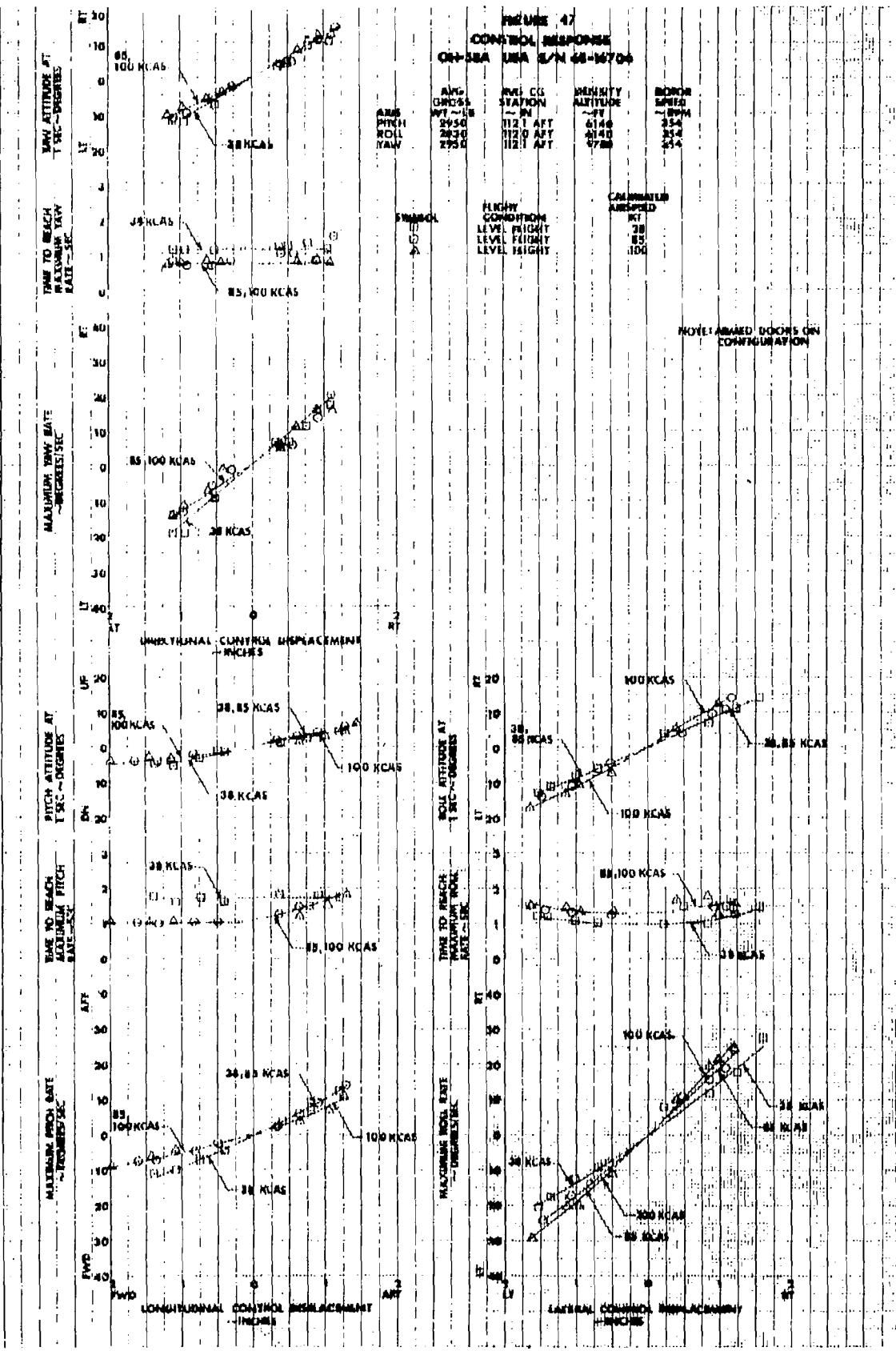
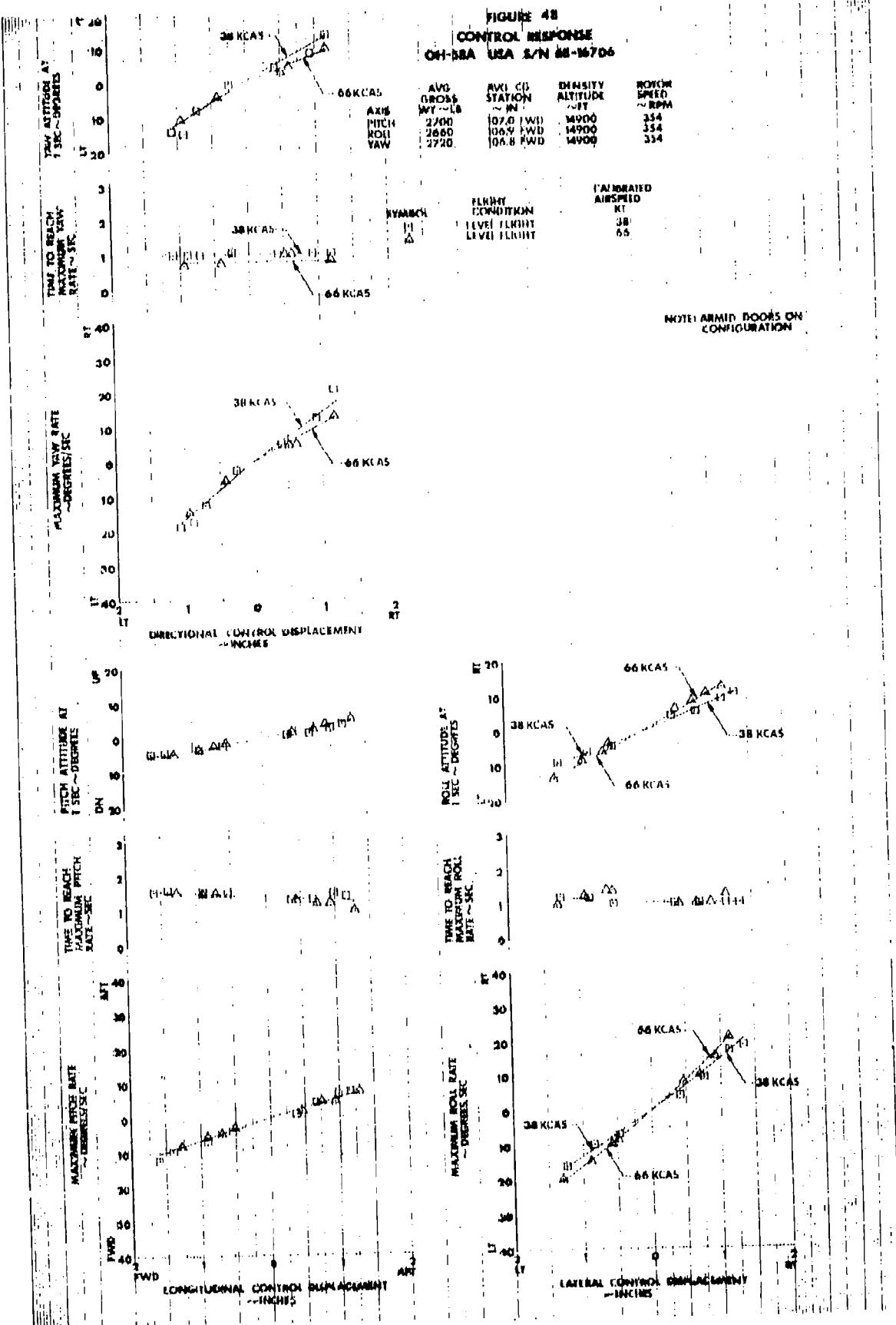


FIGURE 4B  
CONTROL RESPONSE  
OH-58A USA S/N 66-16706



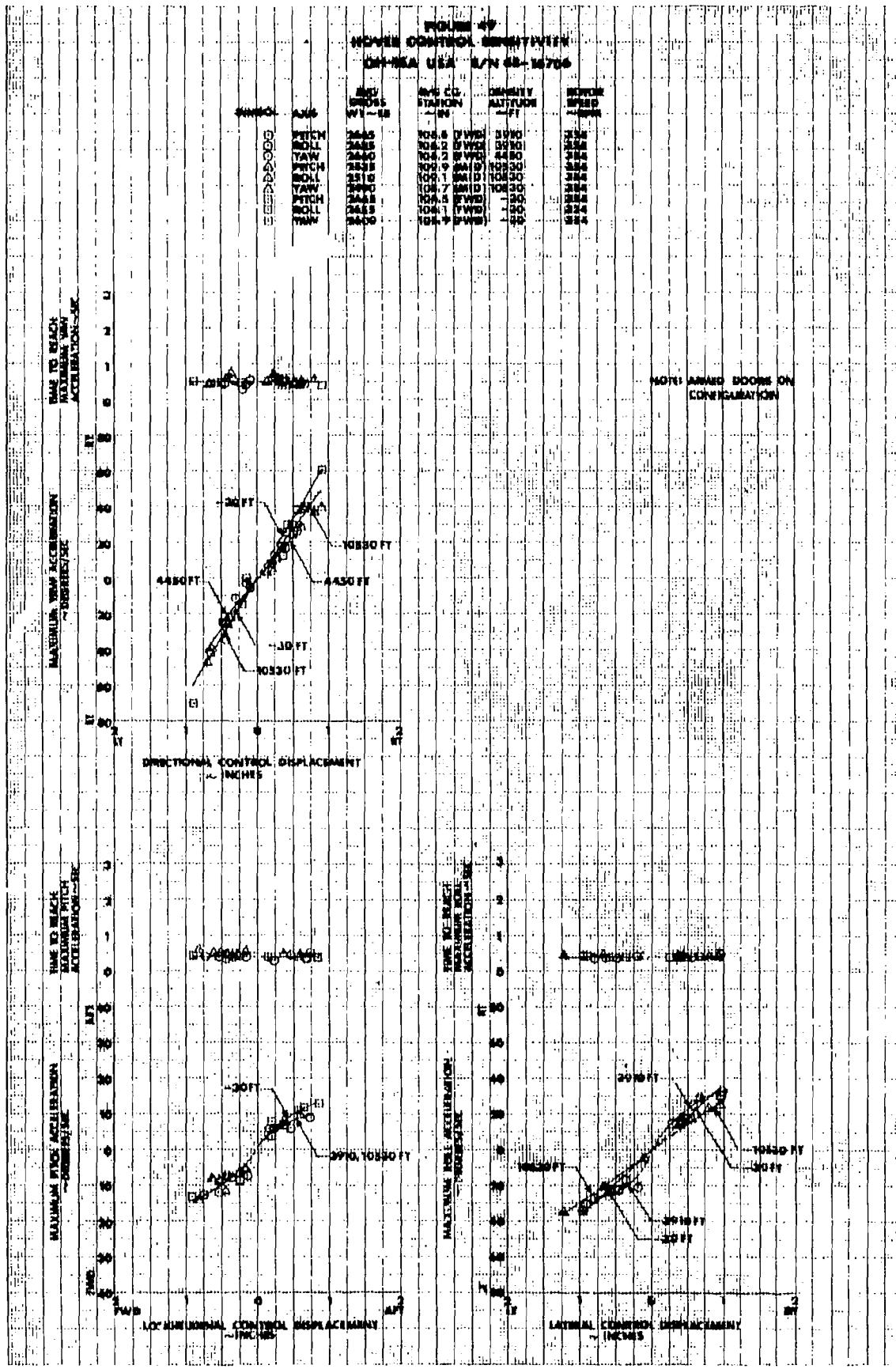


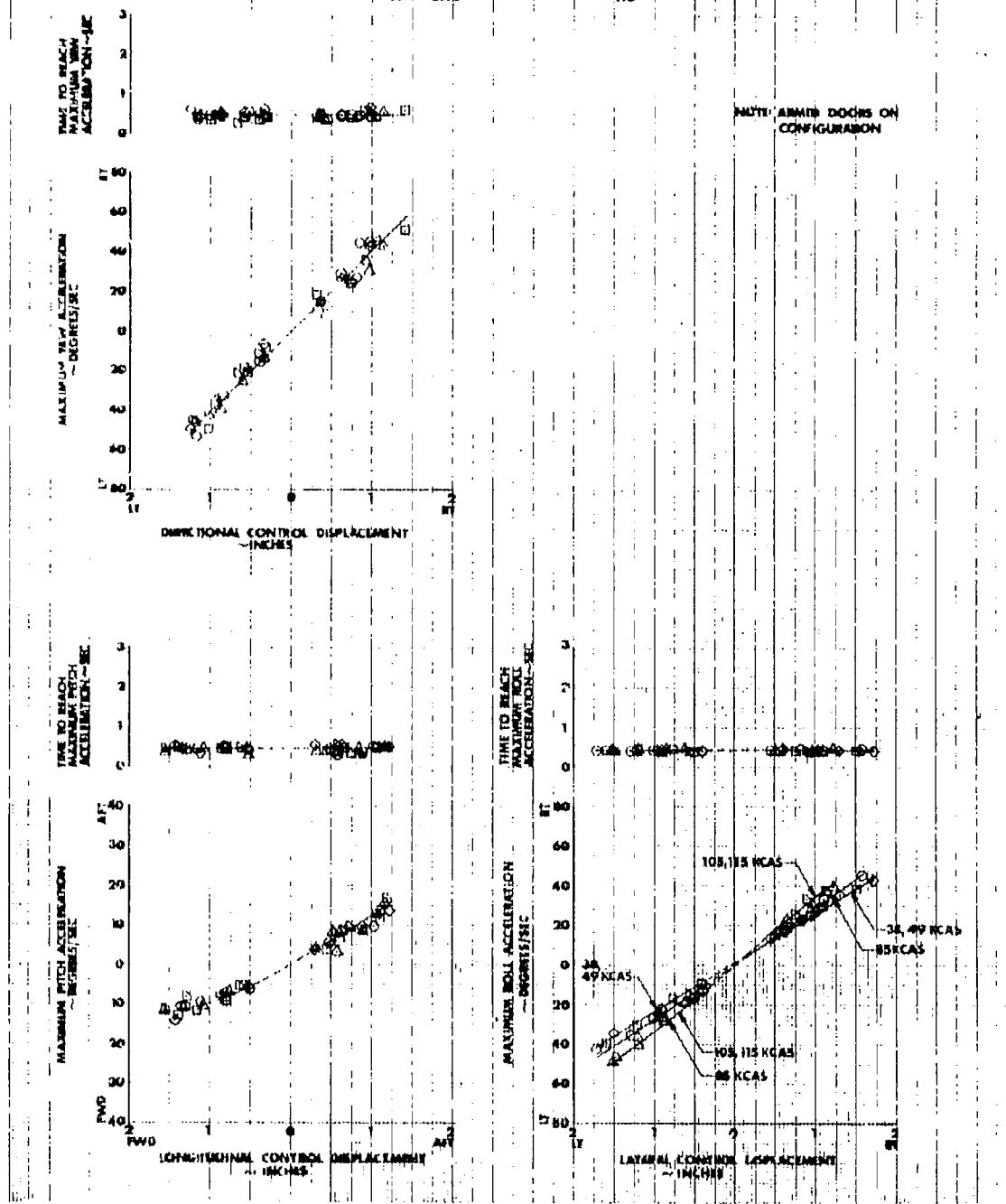
FIGURE 50  
CONTROLS SENSITIVITY  
ON-SEA USA LM 68-16704

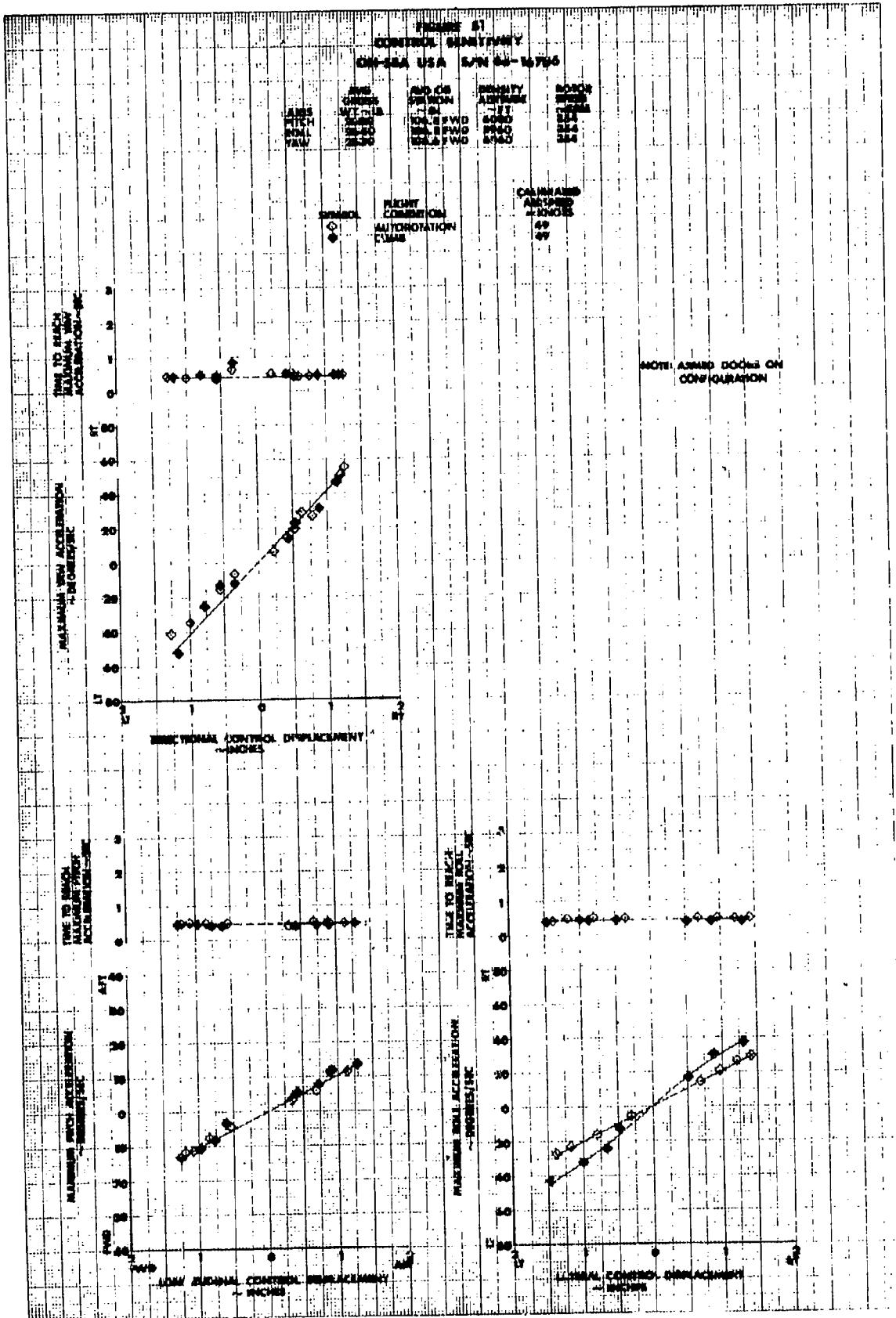
Avg. GROSS WT	Avg. CR SECTION	DENSITY	ROTOR SPEED
24700	1-EN	6000	3200
27000	207.0 FWD	6000	3200
28800	100.0 FWD	6000	3200

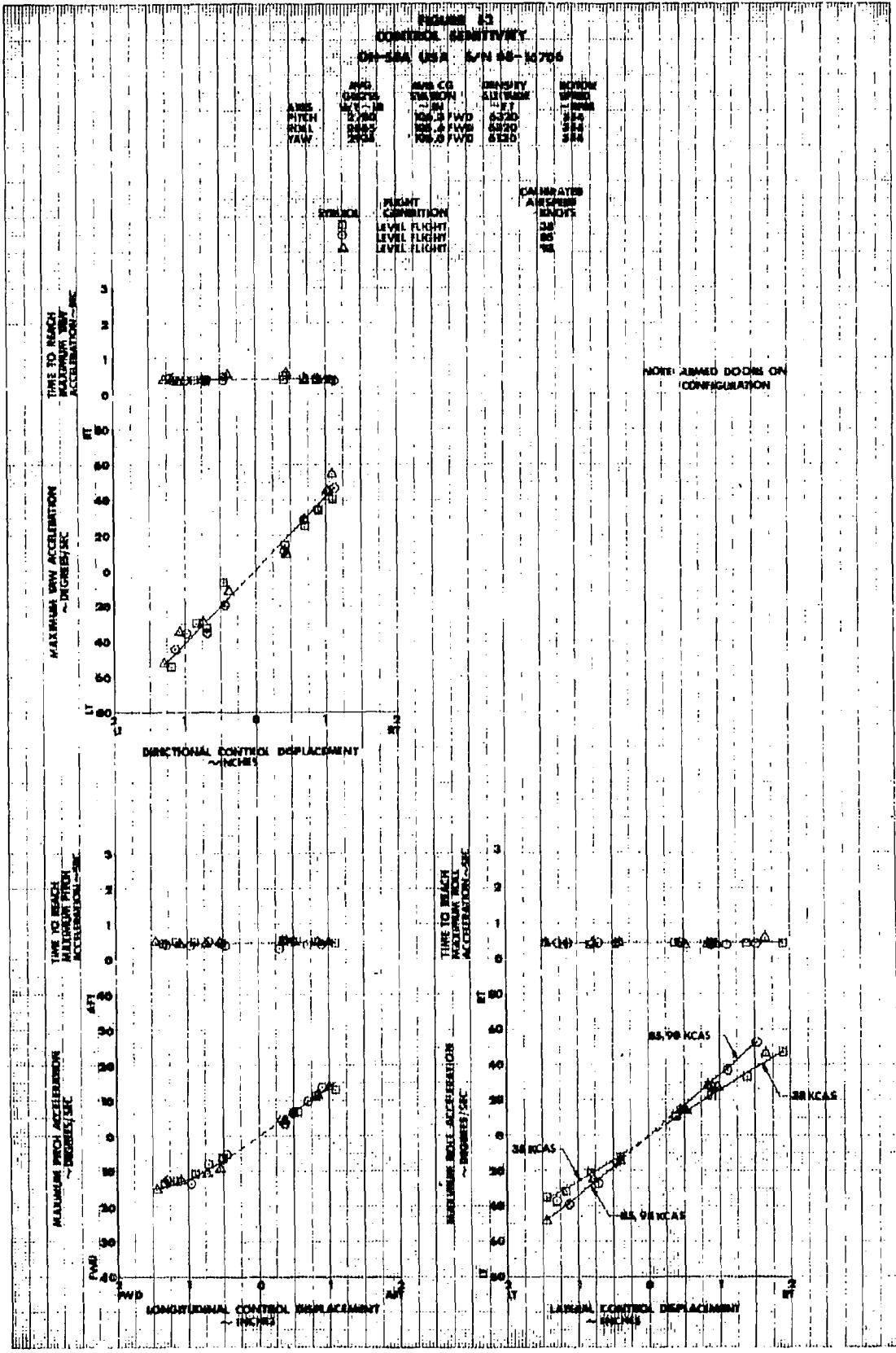
SYMBOL  
DDE-3  
FLIGHT  
CONDITION  
LEVEL FLIGHT  
LEVEL FLIGHT  
LEVEL FLIGHT  
LEVEL FLIGHT  
DIVE

CALM AT 1000  
AIRSPD  
KNOTS  
120  
130  
135  
140  
145  
150  
155

MULTI AMMO DOORS ON  
CONFIGURATION







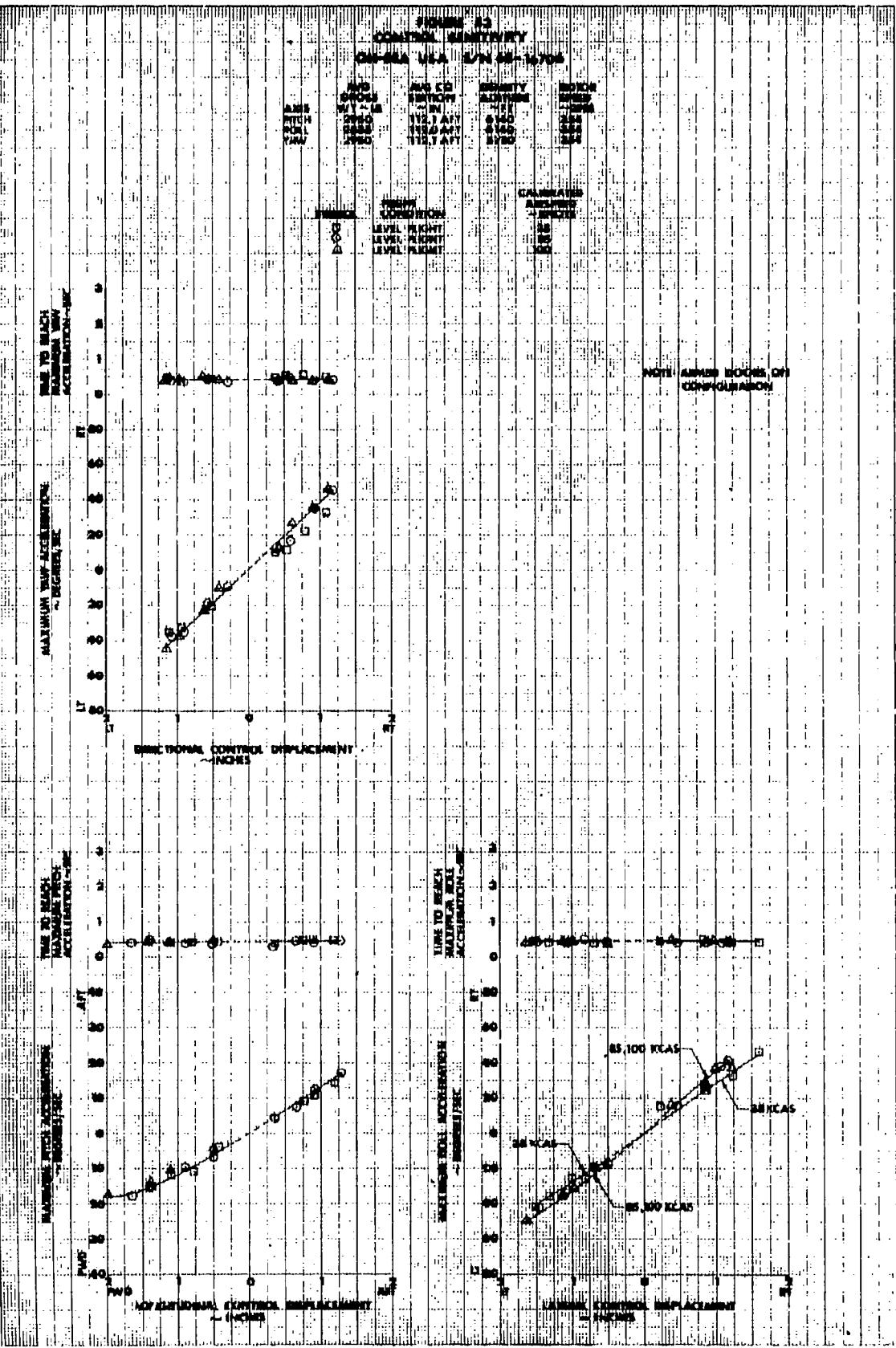


FIGURE 14  
CONTROLS SENSITIVITY

OM-SEA USA S/N 68-16706

WEIGHT	Avg Gross Weight 24,200	WING CO. STATION	DECKING ALTITUDE	HULL SPEED
ROLL	107.0 FWD	14000	20000	20000
YAW	104.8 FWD	14000	20000	20000

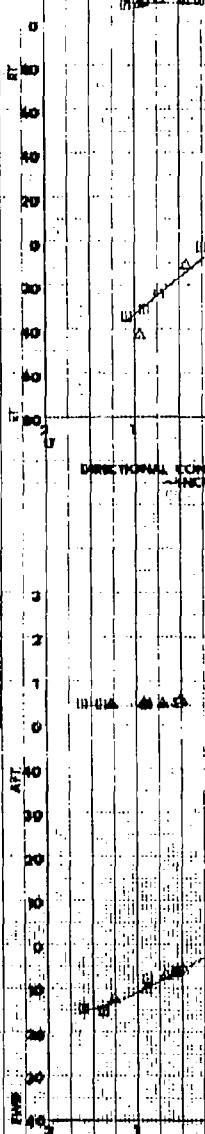
UNBALD  
FLIGHT  
CONDITION  
LEVEL FLIGHT  
LEVEL FLIGHT

CAMERATED  
ANGLE OF  
ATTITUDE  
10000'  
15000'  
20000'

NOTE: ARMED DOORS ON  
DOWNHILL SIDE

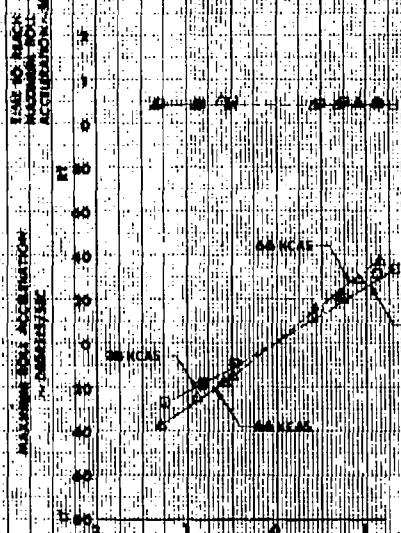
MAXIMUM PITCH ACCELERATION  
TIME TO REACH  
MAXIMUM PITCH  
ACCELERATION SEC

MAXIMUM PITCH ACCELERATION  
TIME TO REACH  
MAXIMUM PITCH  
ACCELERATION SEC



DIRECTIONAL CONTROL DISPLACEMENT  
INCHES

MAXIMUM DIRECTIONAL  
ACCELERATION  
TIME TO REACH  
MAXIMUM DIRECTIONAL  
ACCELERATION SEC



LATERAL CONTROL DISPLACEMENT  
INCHES

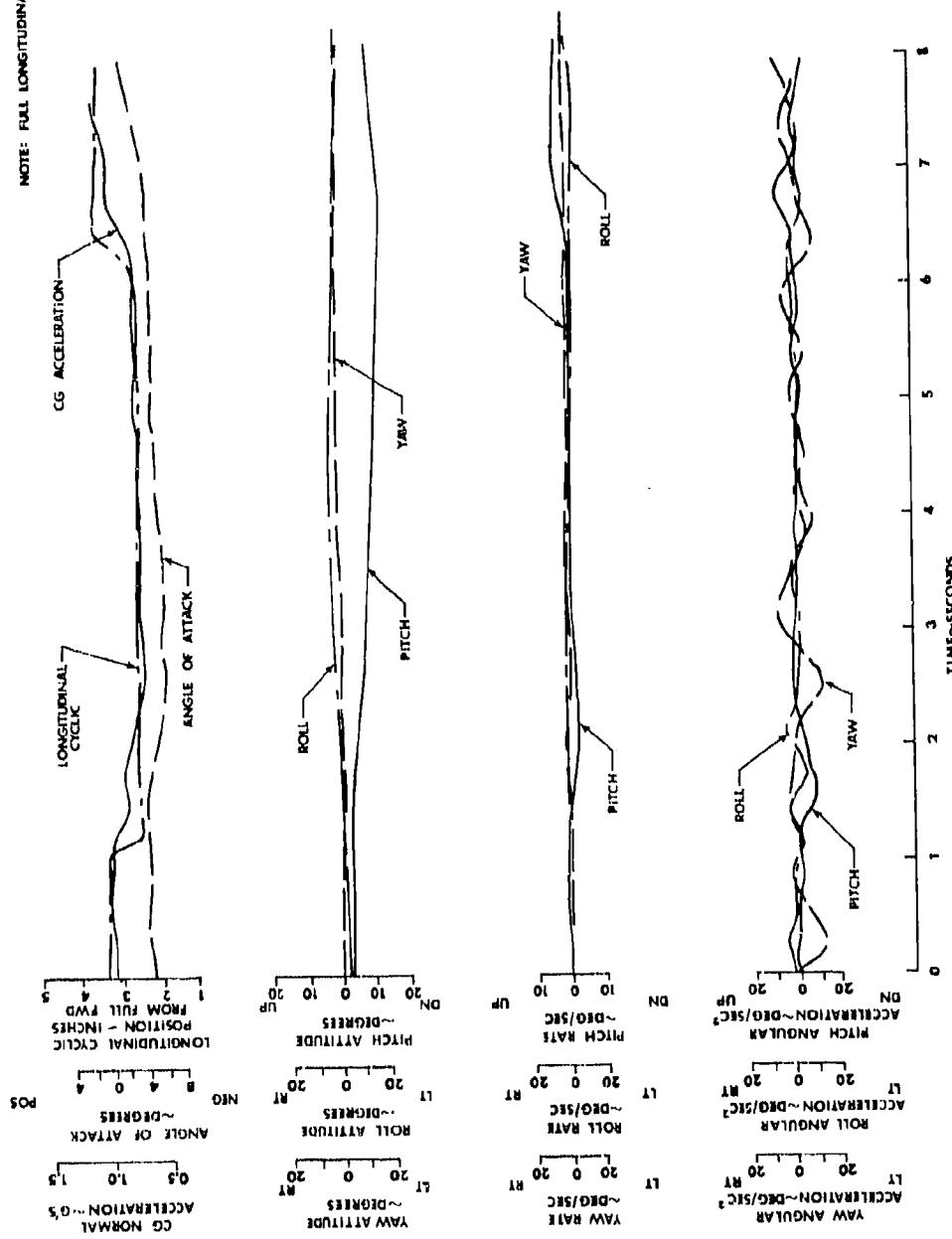
FIGURE 55  
LONGITUDINAL STEP IN LEVEL FLIGHT

OH-58A USA S/N 68-16706

DENSITY ALTITUDE ~ 6080 FT  
CALIBRATED AIRSPEED ~ 95 KTS  
ROTAR SPEED ~ 354 RPM

GROSS WEIGHT ~ 2680 LB  
CG POSITION ~ 107.0 IN (FWD)  
CONFIGURATION ~ ARMED (DOSES ON)

NOTE: FULL LONGITUDINAL CONTROL TRAVEL = 12.00 IN



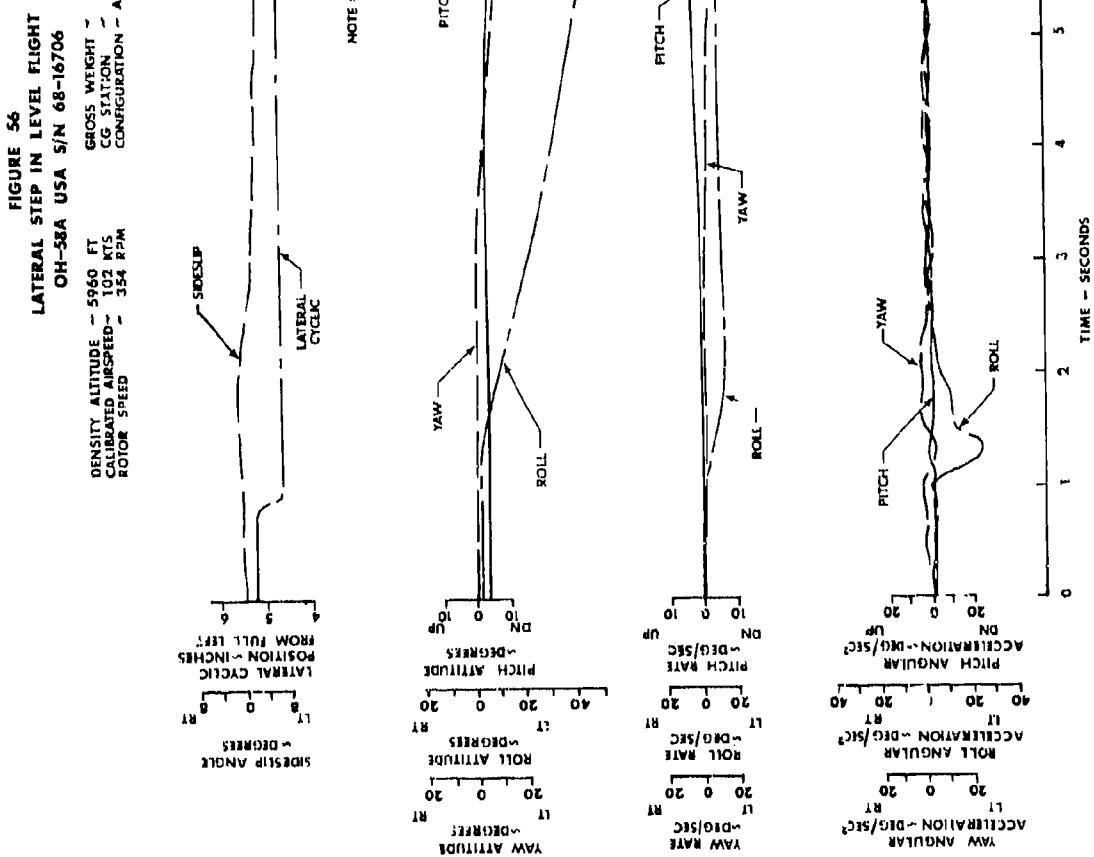
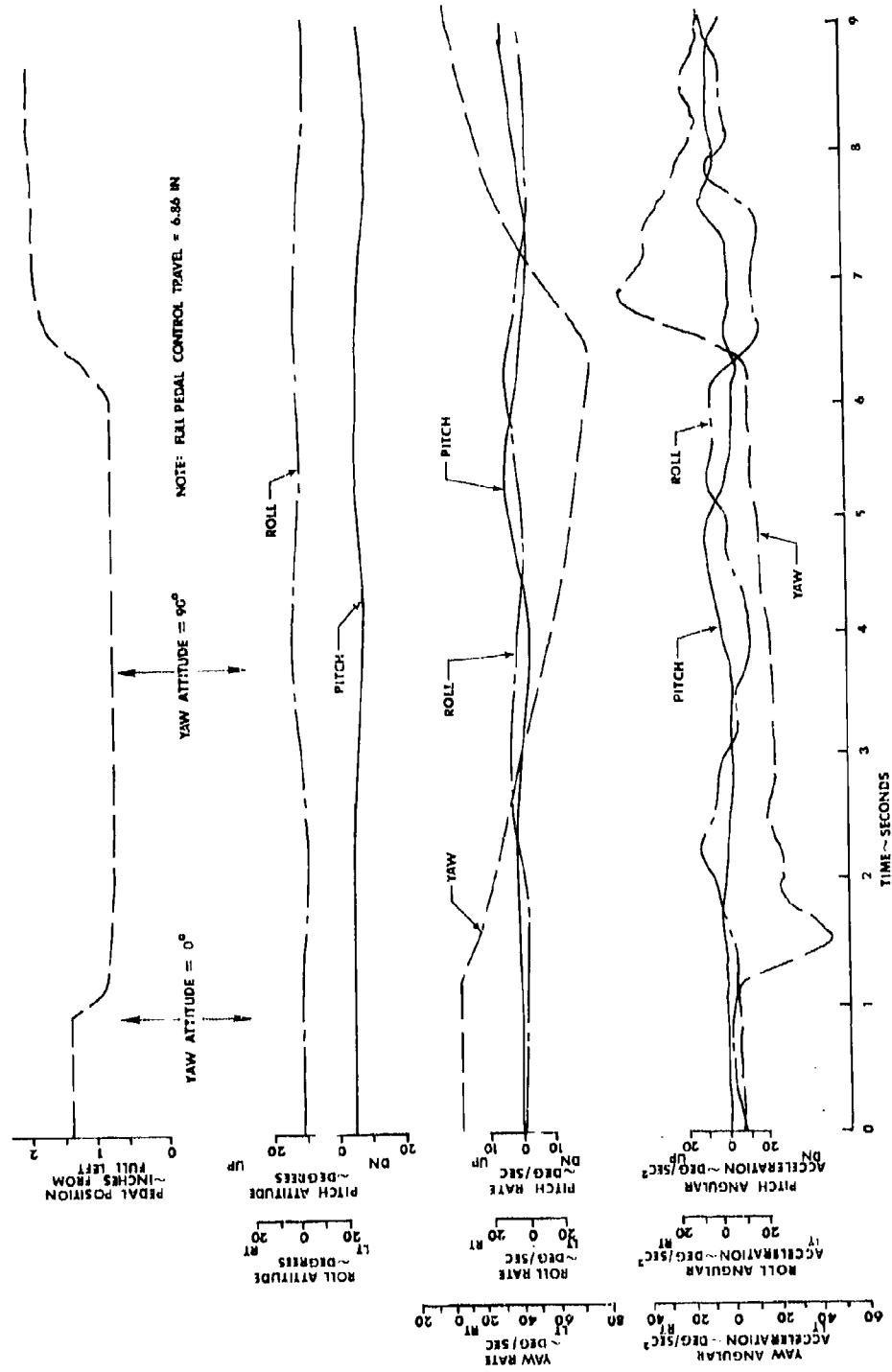


FIGURE 57

RIGHT PEDAL STEP IN HOVER  
 OH-58 USA S/N 68-16706  
 DENSITY ALTITUDE ~ - 940 FT  
 CALIBRATE AIRSPEED ~ HOVER  
 MOTOR SPEED ~ 354 RPM

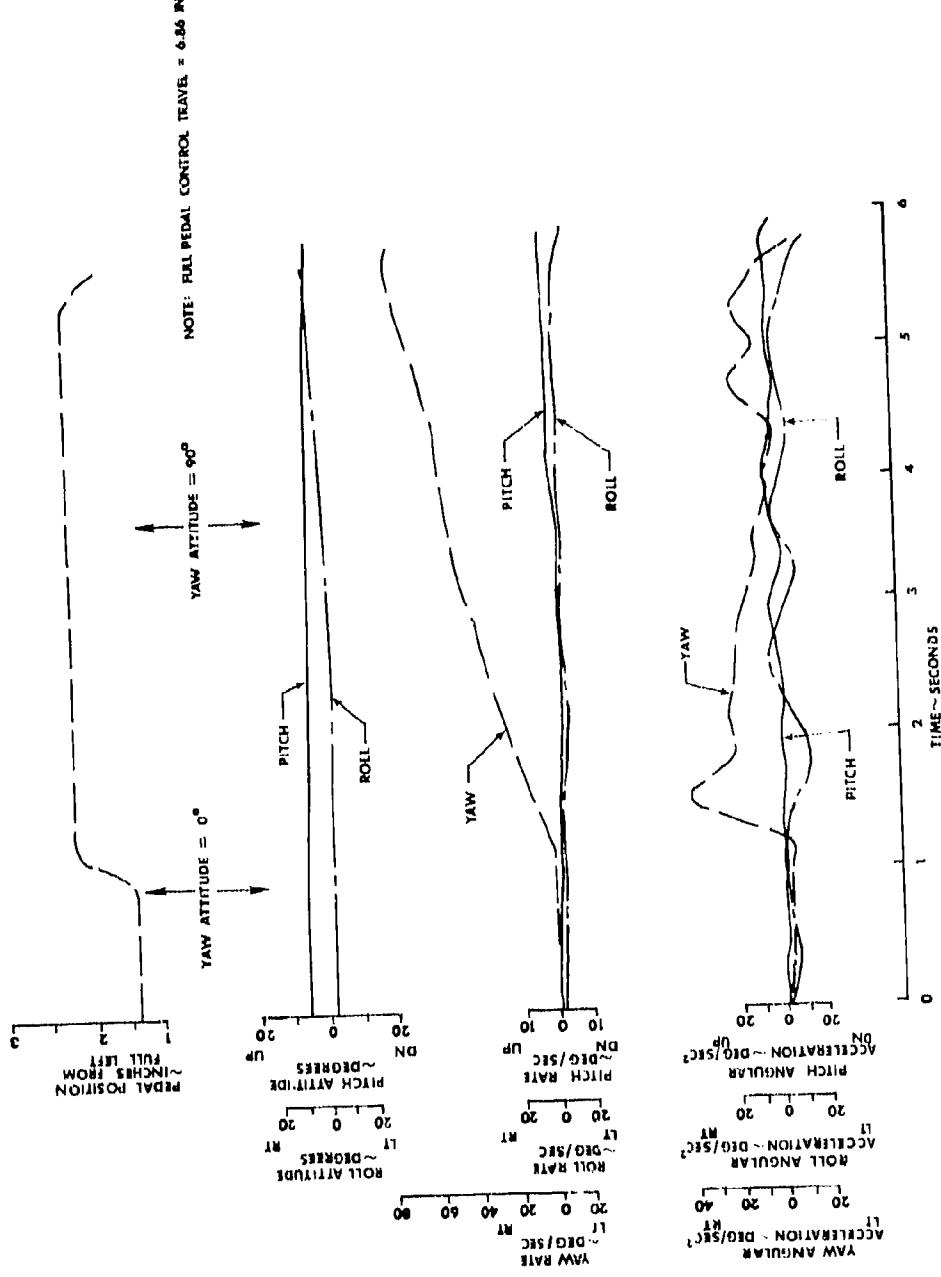
GROSS WEIGHT ~ 3025 LB  
 CG STATION ~ 112.8 IN (AFT)  
 CONFIGURATION ~ ARMED (DOORS ON)



**FIGURE 56**  
**LEFT PEDAL STEP IN HOVER**  
**OH-58A USA S/N 68-36706**

DENSITY ALTITUDE  $\sim$  940 FT  
 CALIBRATE AIRSPEED  $\sim$  HOVER  
 MOTOR SPEED 354 RPM

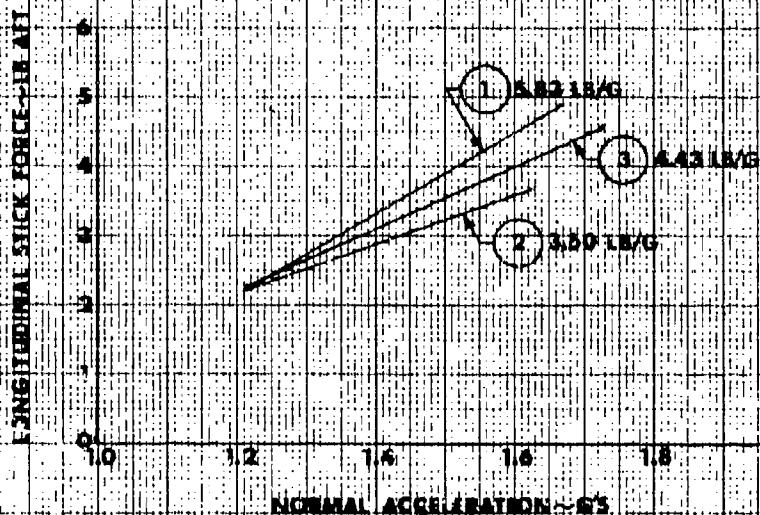
GROSS WEIGHT  $\sim$  3015 LB  
 CG STATION  $\sim$  112.2 IN (A.F.)  
 CONFIGURATION  $\sim$  ARMED (DOORS ON)

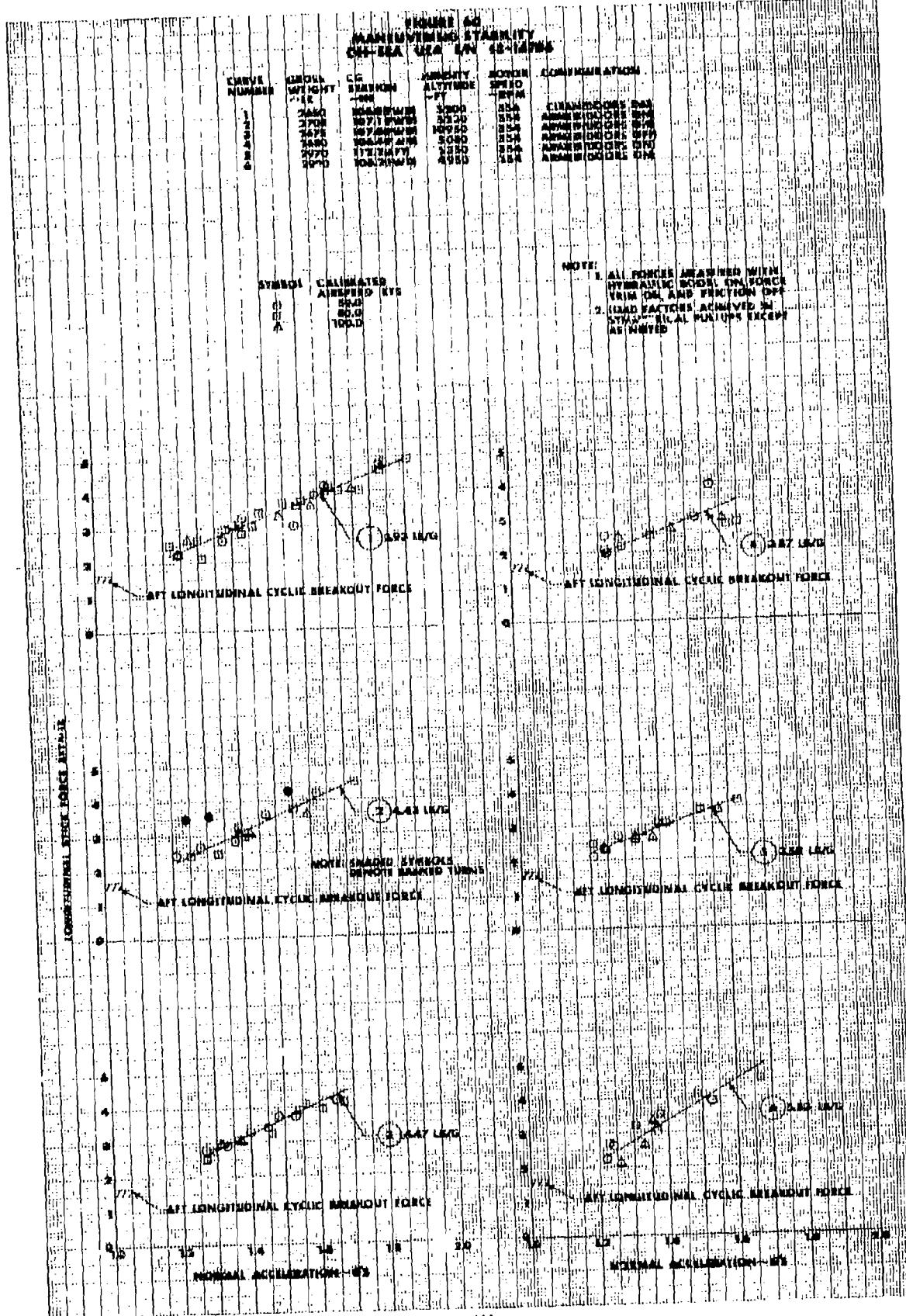


**MANEUVERING STABILITY SUMMARY**  
**CH-53A USA S/N 68-16706**

CURVE NUMBER	GROSS WEIGHT - LB	CG STATION	DENSITY	ALTITUDE	ROTOR SPEED - RPM	CONFIGURATION		
						ARMED (DOORS ON)	ARMED (DOORS ON)	ARMED (DOORS ON)
1	2900	106.2 (FW/D)	4050	0	324			
2	2930	112.9 (AFT)	5250	0	324			
3	2705	107.1 (FW/D)	5220	0	324			

NOTE: FOR AIRSPEED RANGE  
 FROM 59 KCAS TO 100 KCAS





**FIGURE 61**  
**MANEUVERING STABILITY IN SYMMETRICAL PULL-UP**  
**OH-58A USA S/N 68-16706**

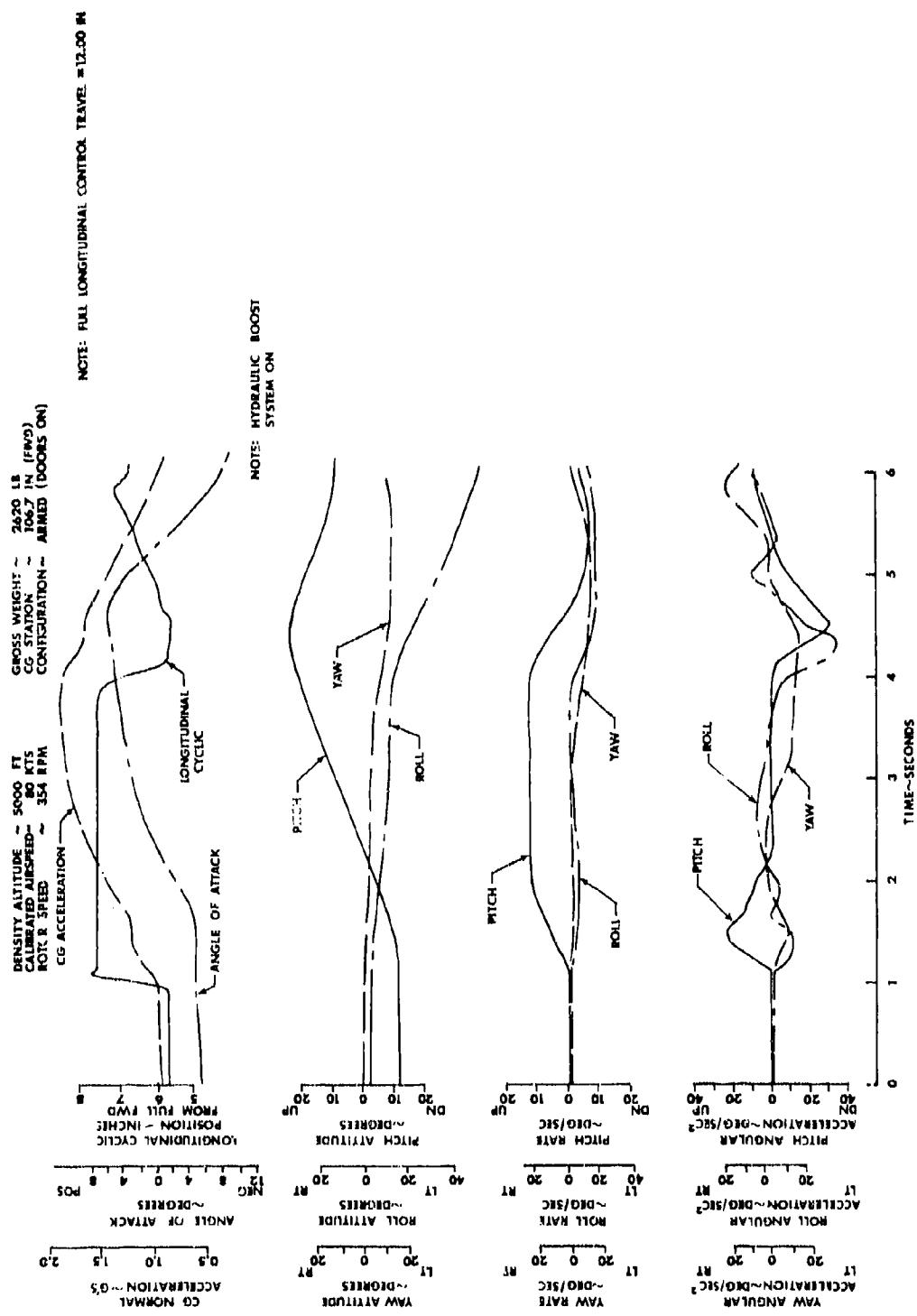
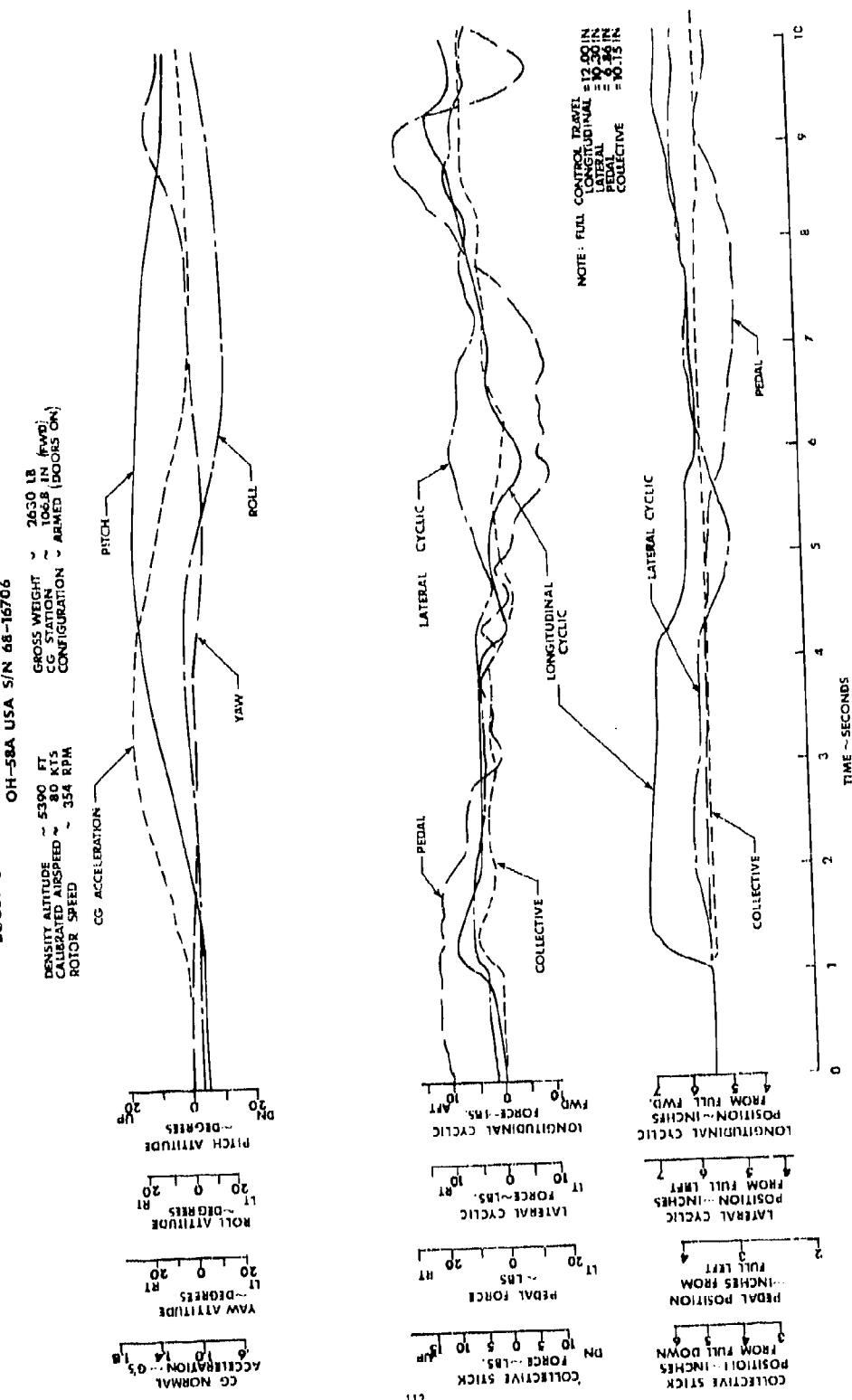


FIGURE 62  
BOOST-OFF MANEUVERING STABILITY IN SYMMETRICAL PULL-UP



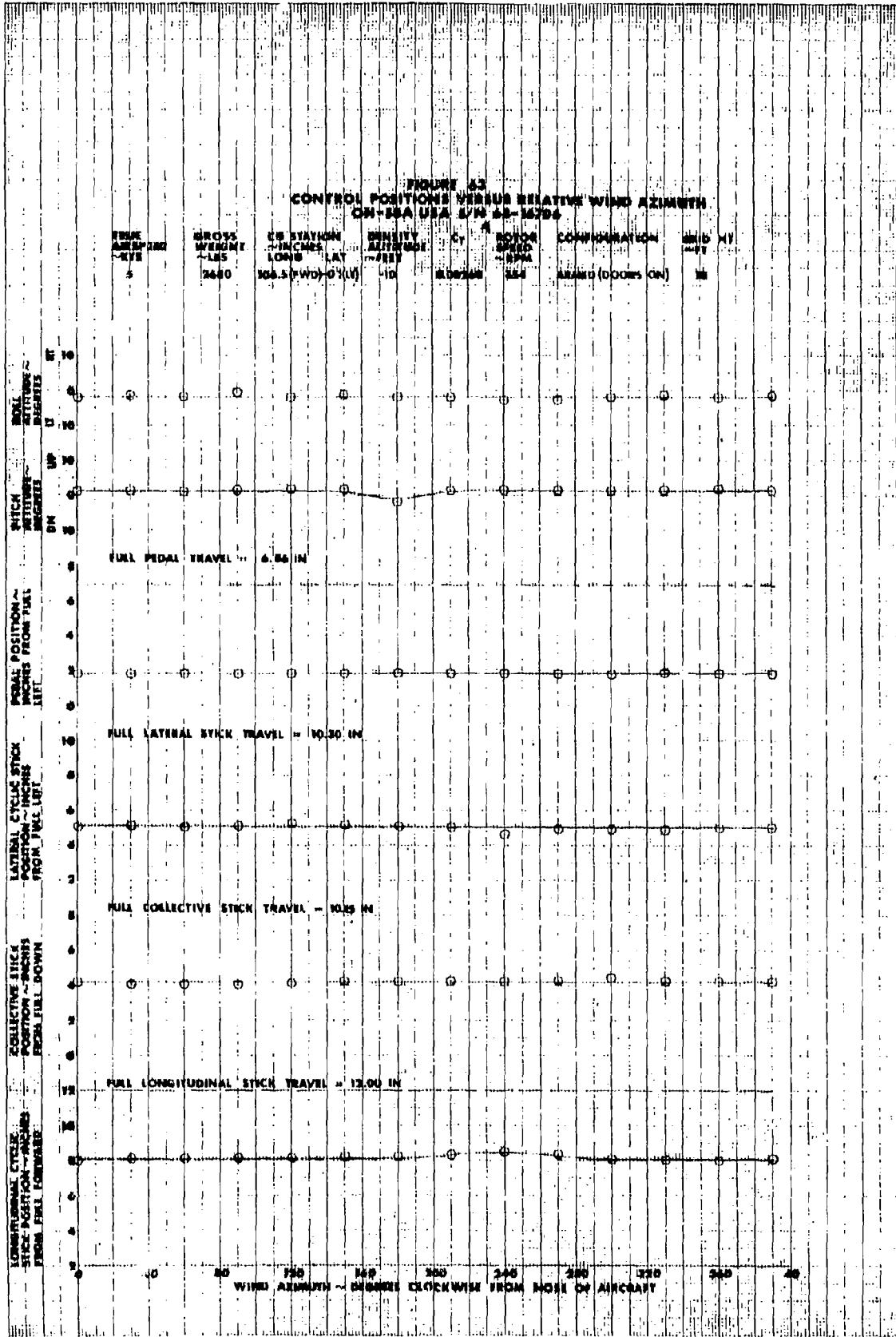
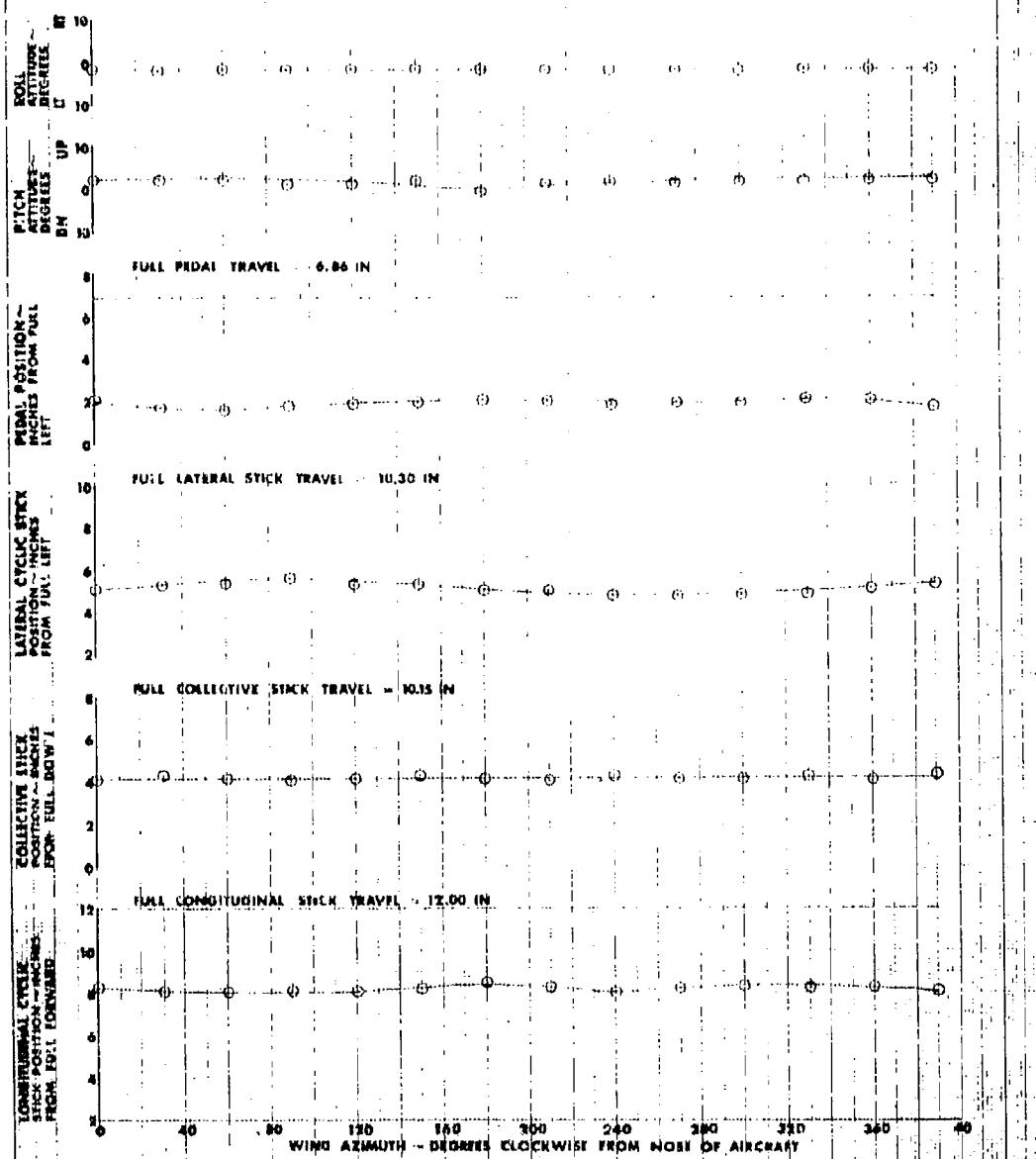


FIGURE 64  
CONTROL POSITIONS VERSUS RELATIVE WIND AZIMUTH  
CH-53A USA S/N 68-10706

TRUE AIRSPEED KTS	GROSS WEIGHT LBS	CG STATION INCHES LONG LAT 106.5(FWD)-0.0(LT)	DENSITY ALTITUDE FEET	G	ROTOR SPEED RPM	CONFIGURATION	SKID HT OFF
10	2680		-10	0.00268	354	ARMED (DOORS ON)	10



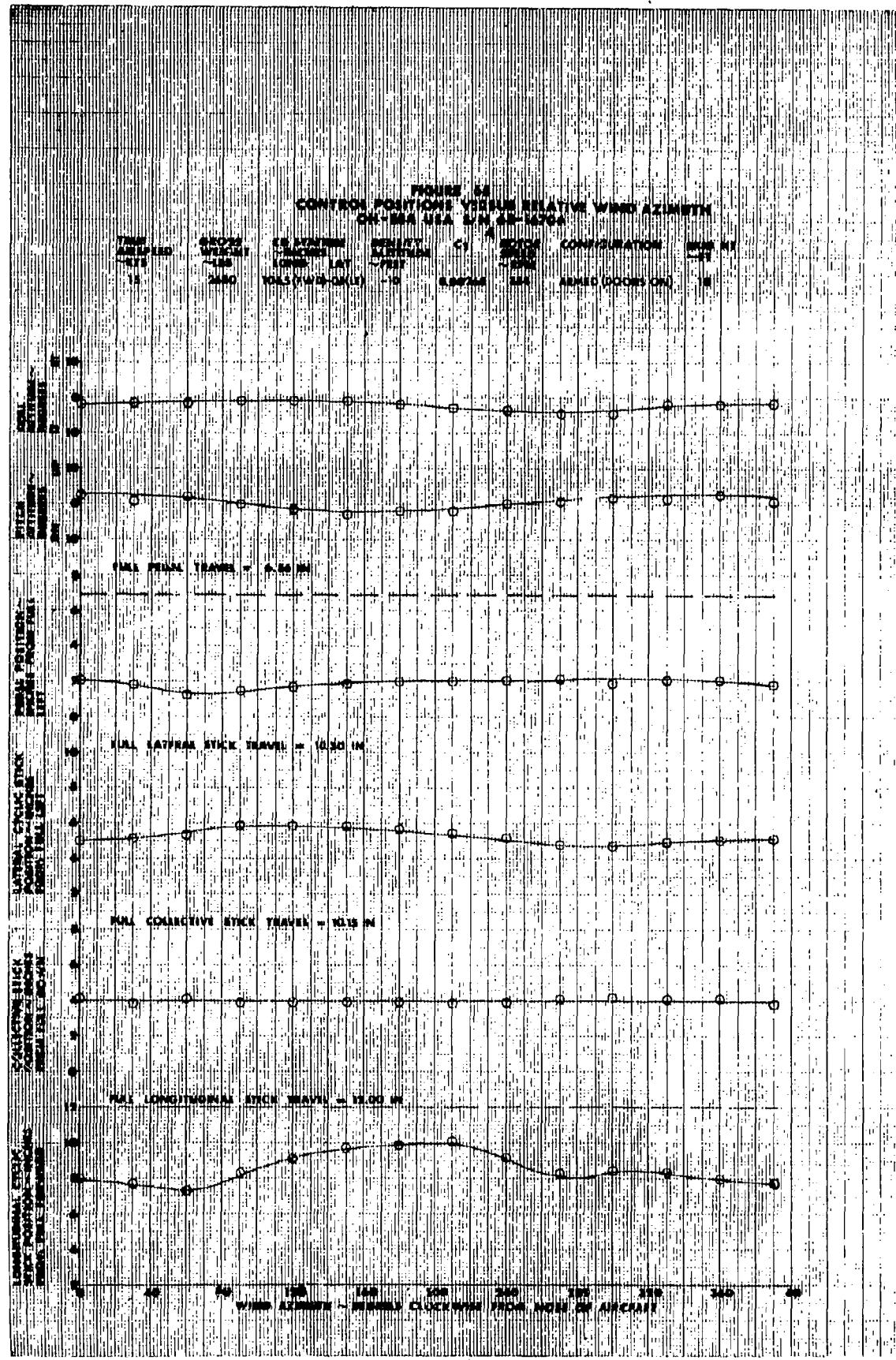
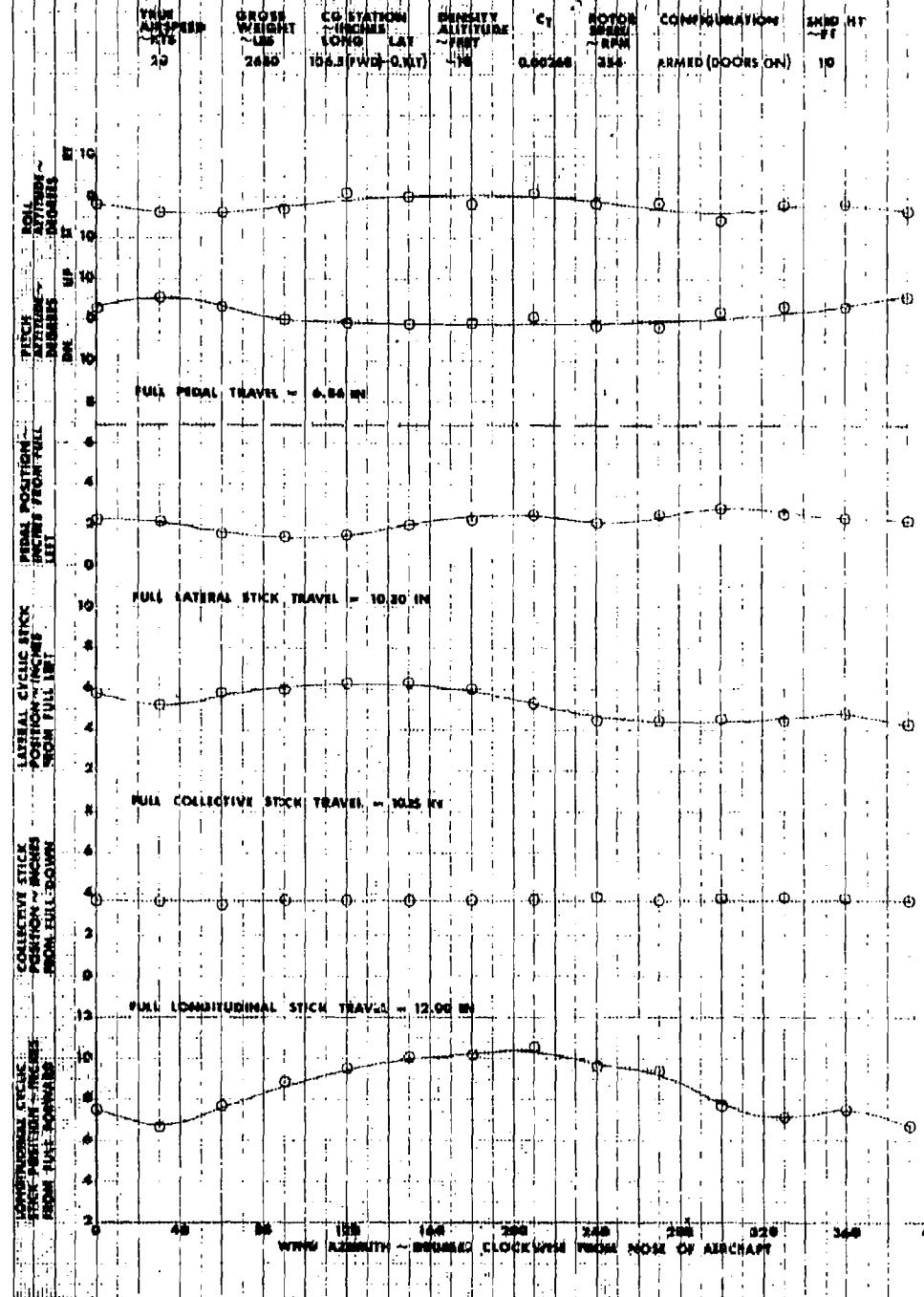
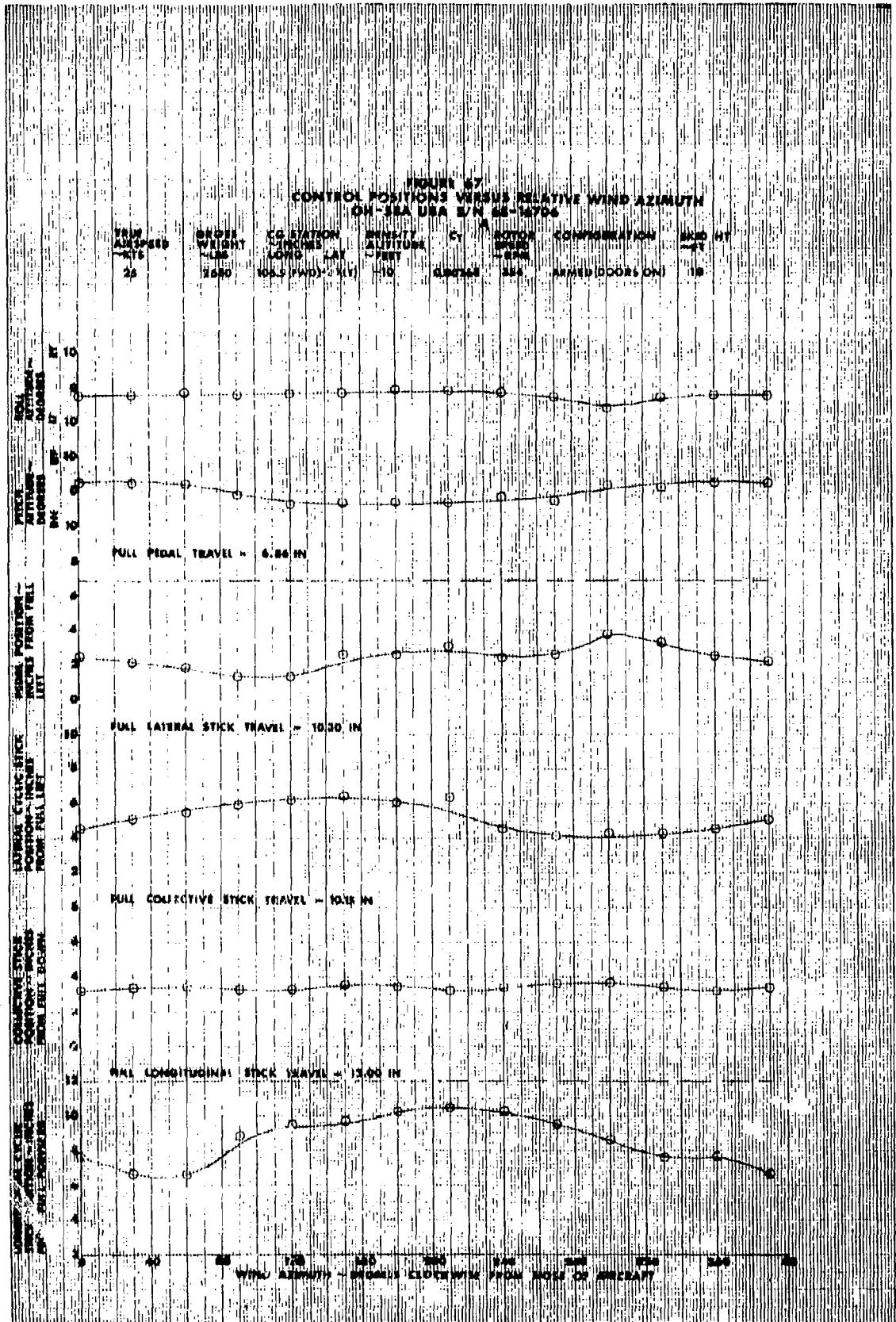
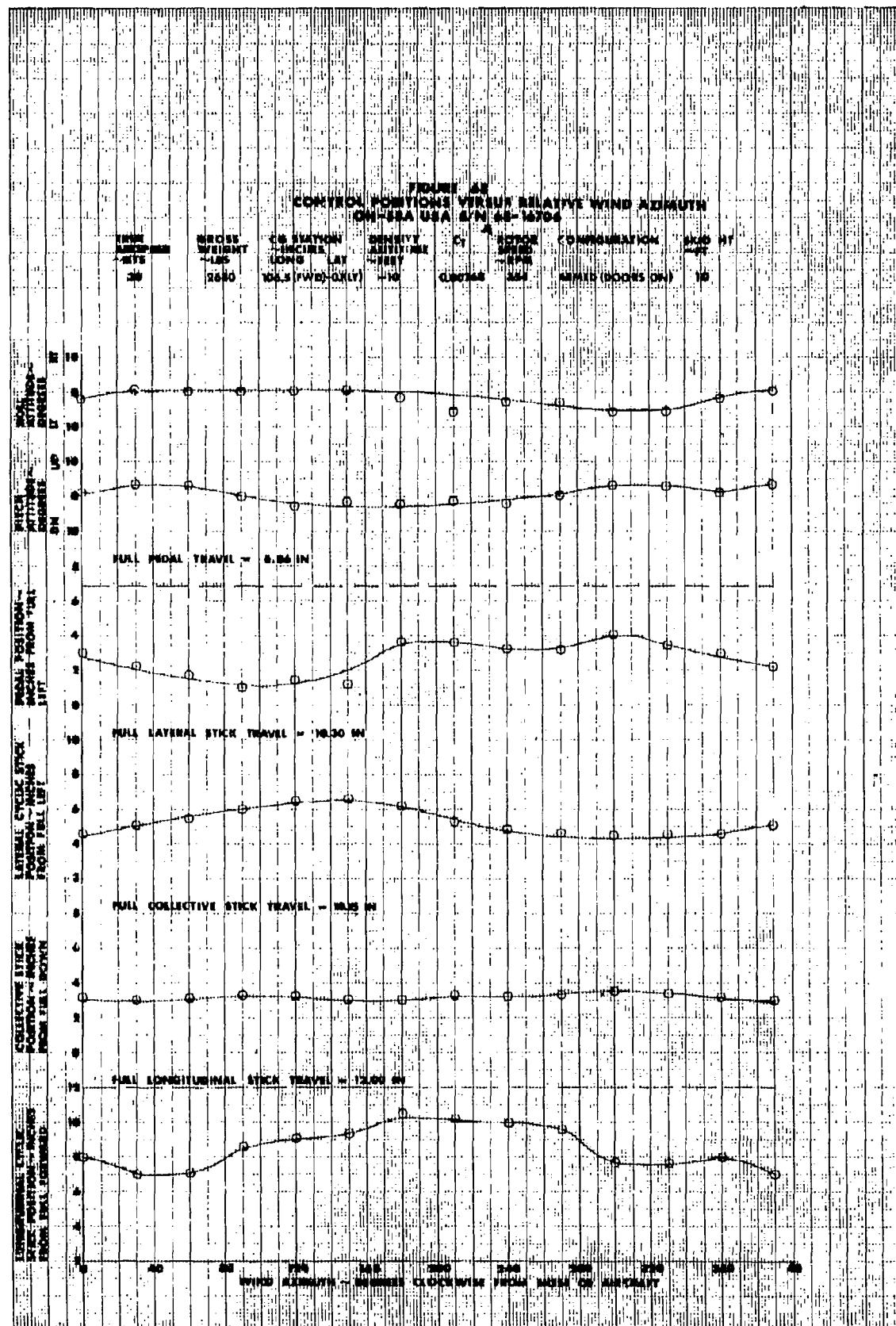


FIGURE 44  
CONTROL POSITIONS VERSUS RELATIVE WIND AZIMUTH







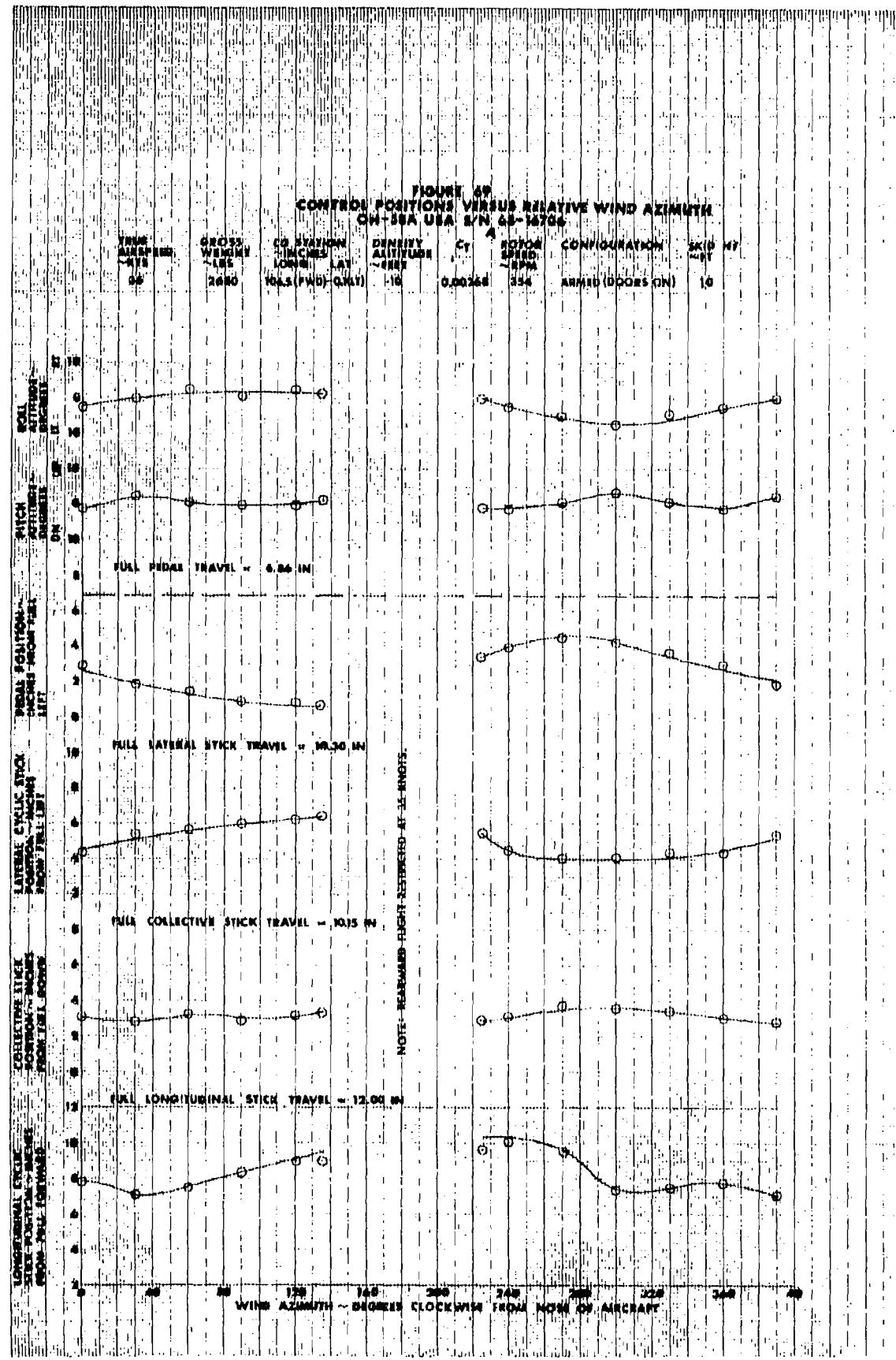
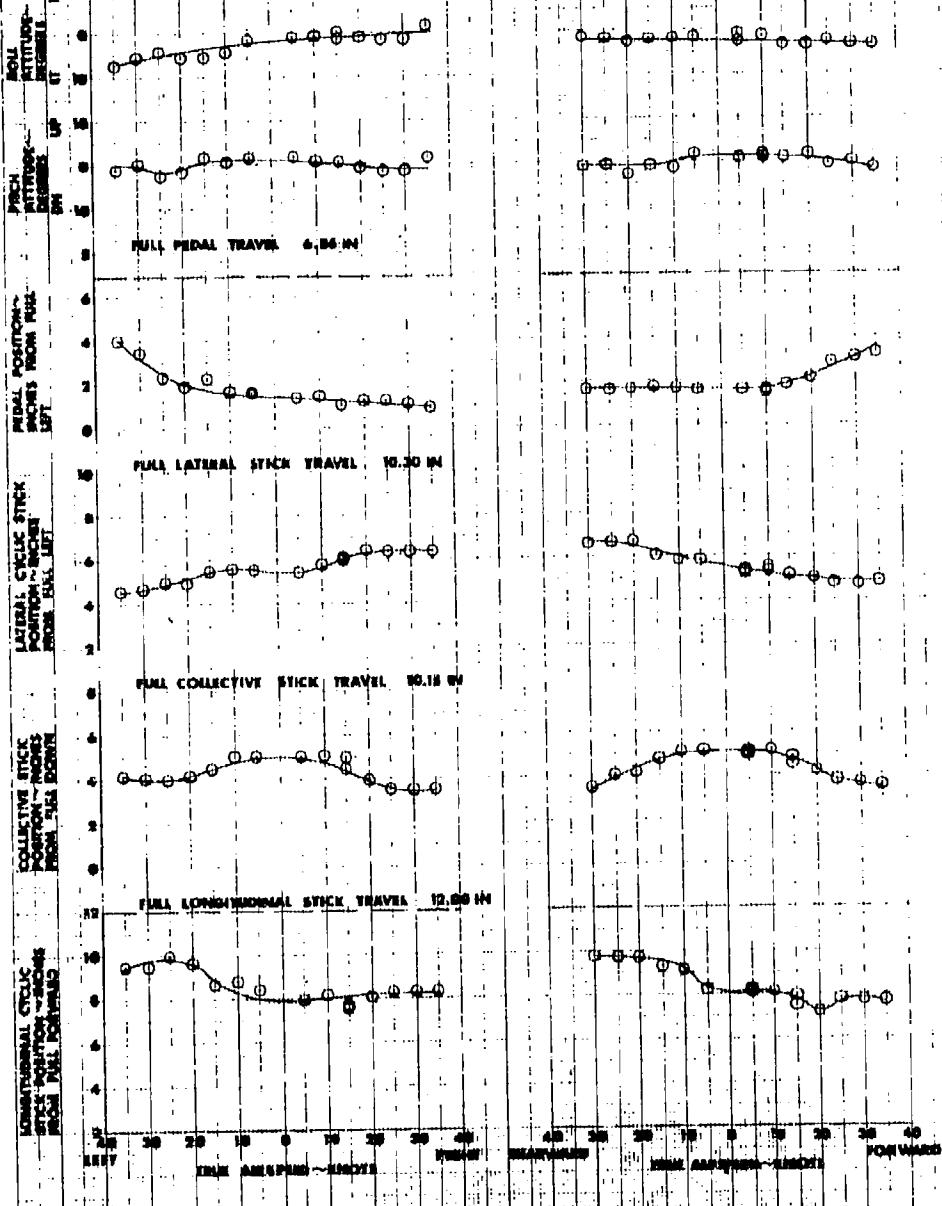


FIGURE 70  
CONTROL POSITIONS IN SIDEWARD, REVERSE AND FORWARD FLIGHT

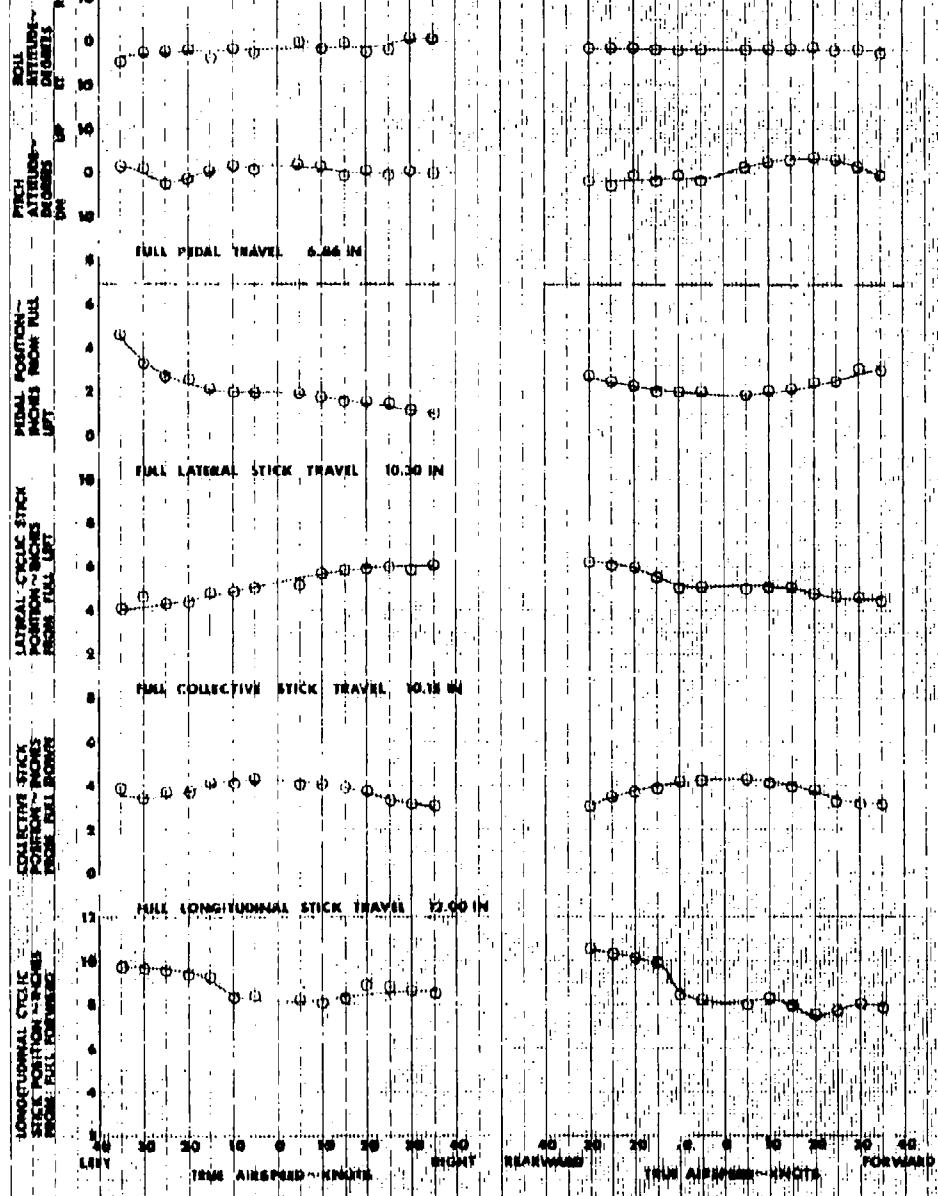
OH-58A USA S/N 68-14786

DROSS C.G. STATION DENSITY C. MOTOR CONFIGURATION 8000 HP  
MOMENT 1.8 IN MRS ALTITUDE 10000 FT SPEED 100 MPH  
2650 LONG. LAT 077° 07° 00' 00" 0.06204 2000 AMPS(DOOR ON) 10



**CONTROL POSITIONS IN SIDEWARD, REARWARD AND FORWARD FLIGHT**  
OH-58A USA C/N 68-14301

GROSS WEIGHT	C. B. STATION	DEPTH	MOTOR SPEED	CONFIGURATION	SHRINK PULLEY
11,000	1000' DEEP	1000'	1000 RPM	1000' DEEP	1000' DEEP



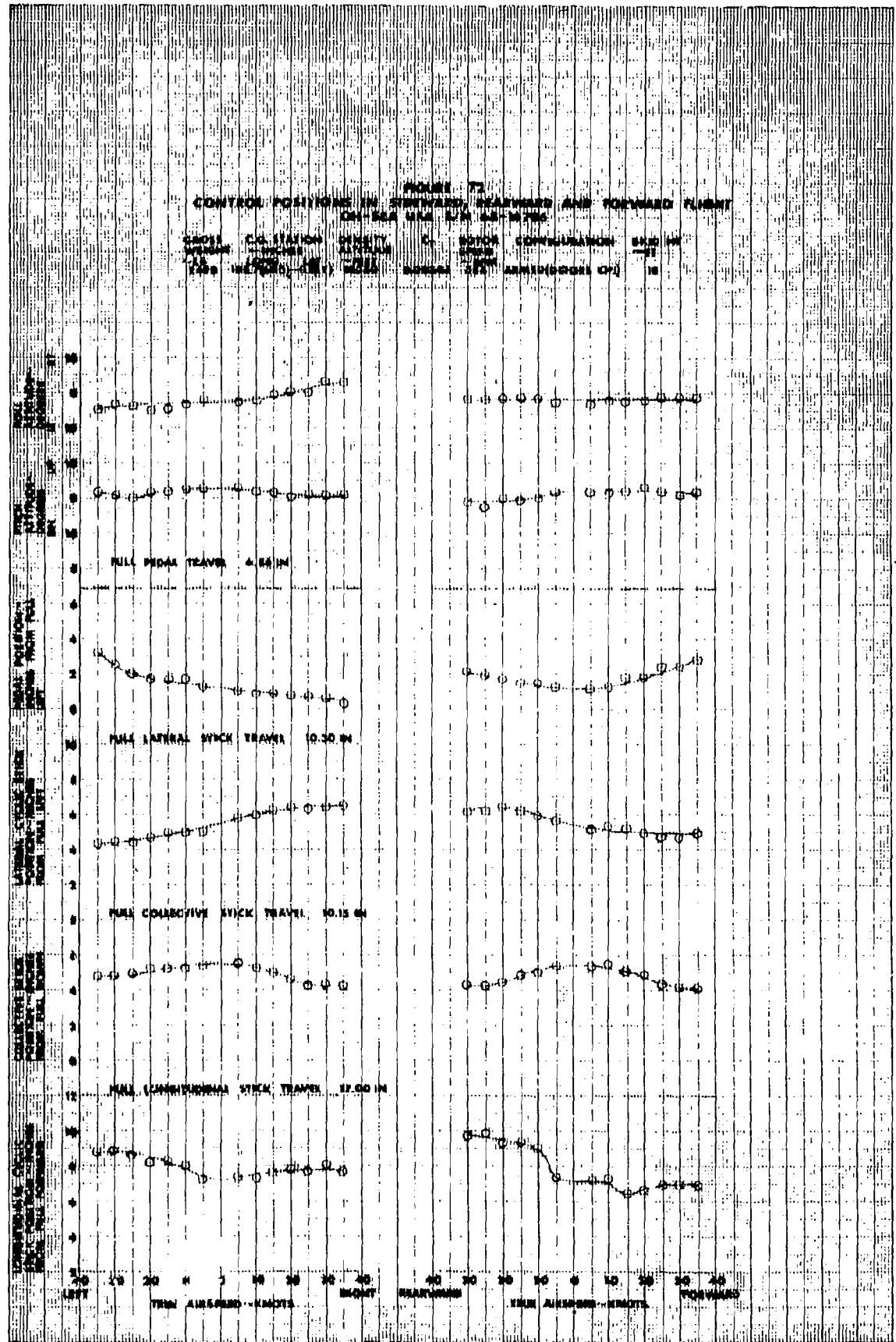


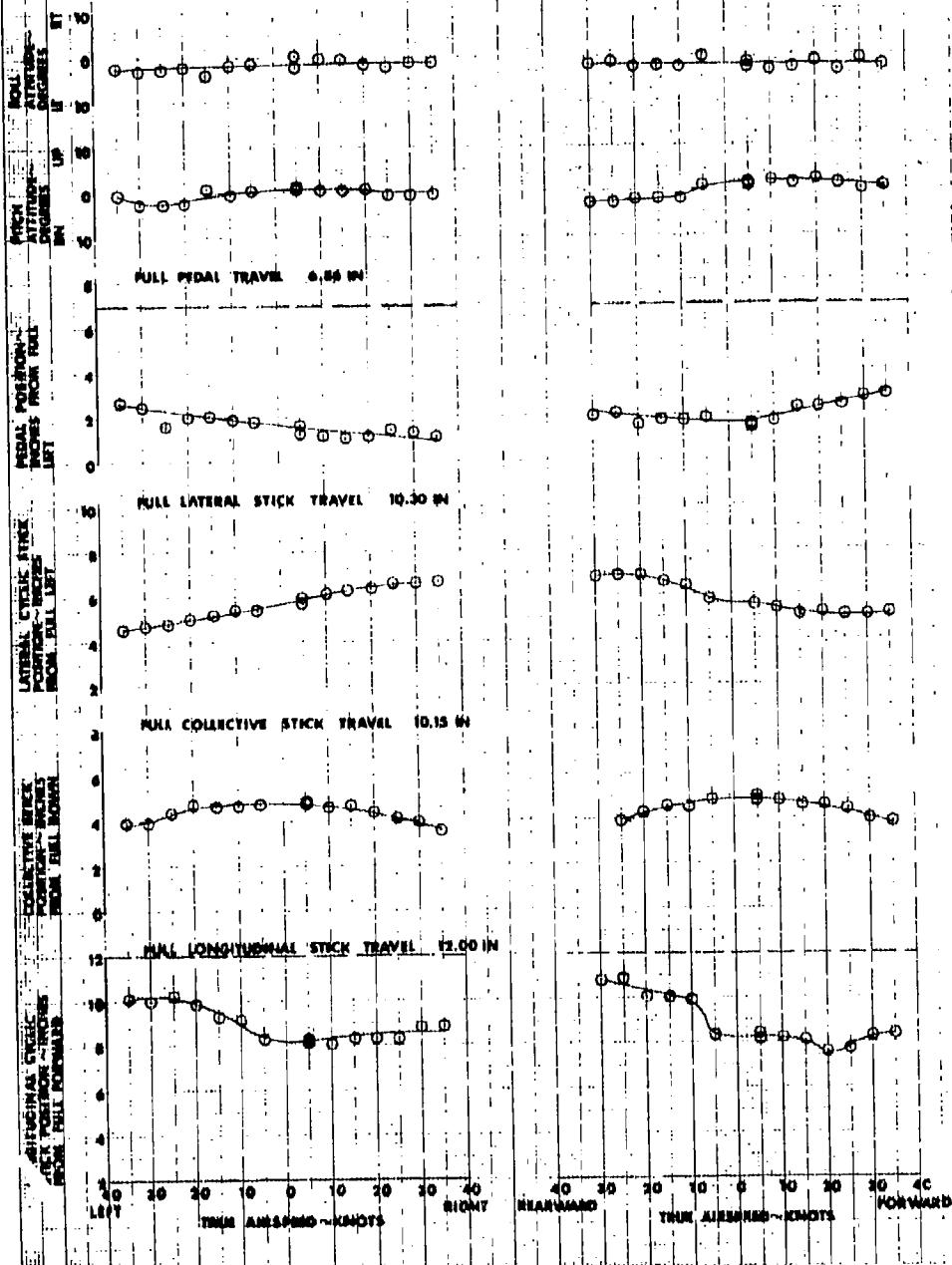
FIGURE 73  
CONTROL POSITIONS IN SIDEWARD, REARWARD AND FORWARD FLIGHT

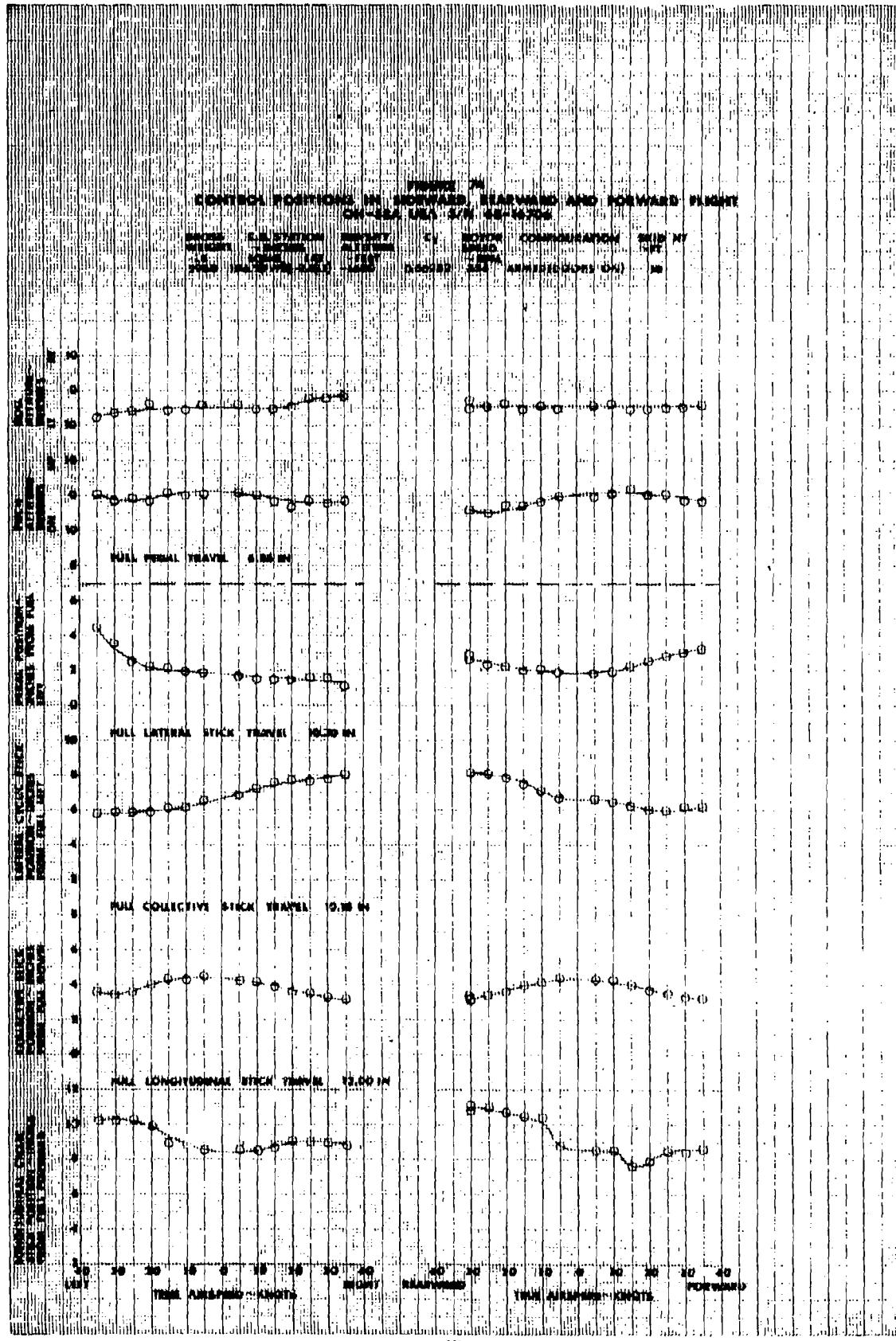
ON-362 USA S/N 68-16766

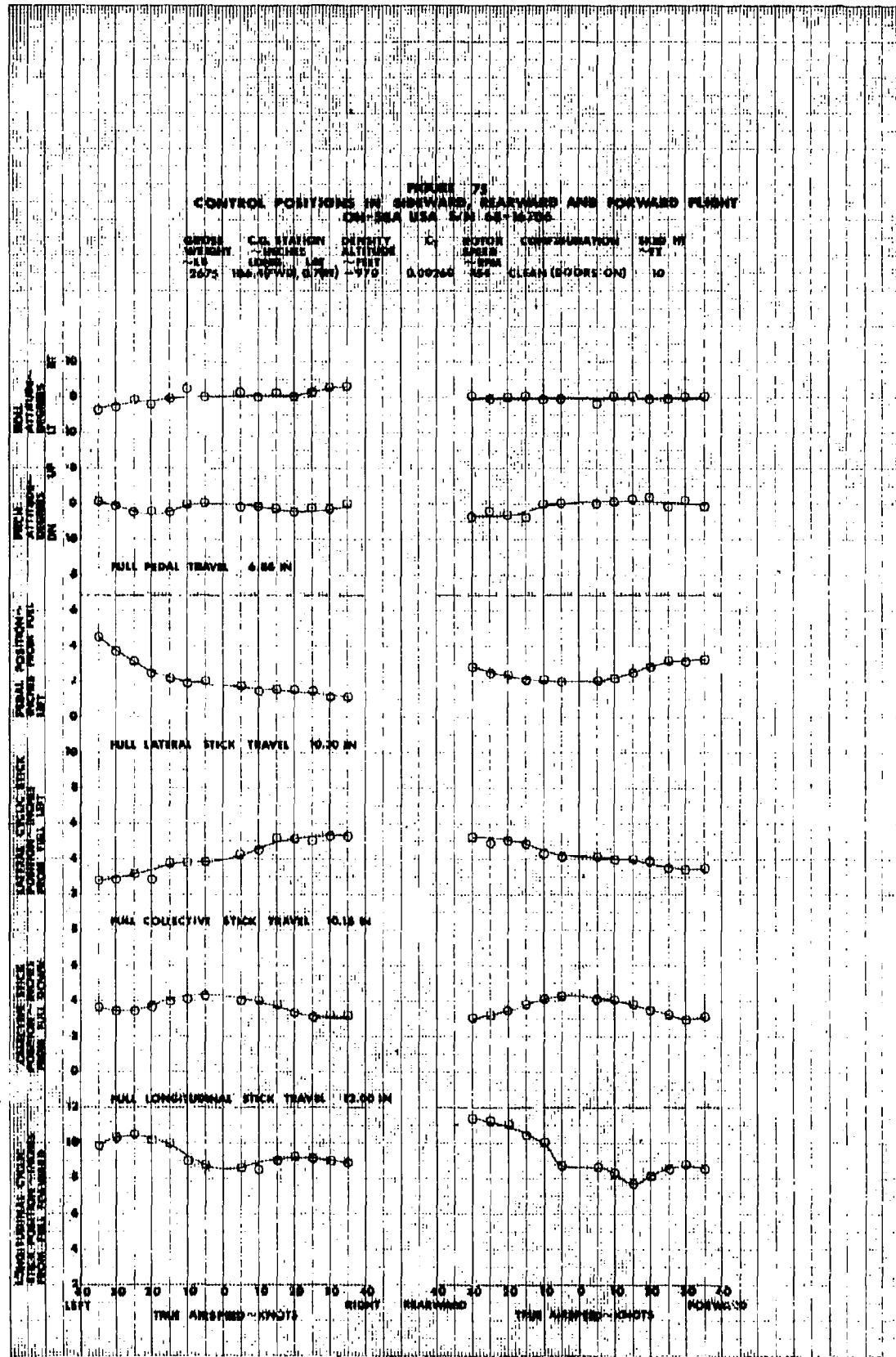
GROSS WEIGHT 2150 LBS C.G. STATION 150 INCHES DENSITY 0.002799 C. MOTOR SPEED 250 RPM CONFIGURATION ARMED DOORS ON

SKID HT

PT







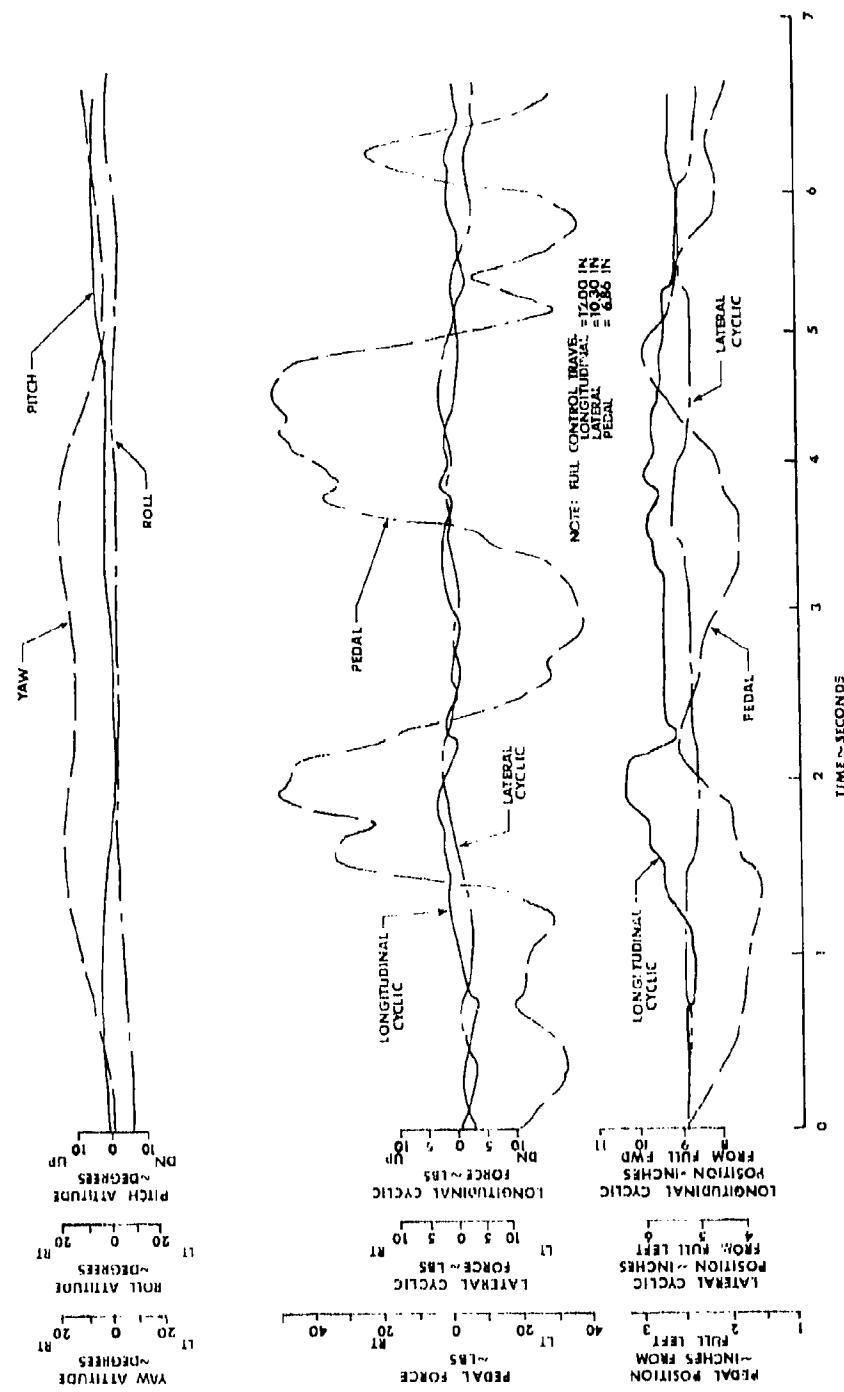
**FIGURE 76**

**LEFT SIDEWARD FLIGHT AT 15 KIAS**

**OH-58A USA SIN 68-16706**

DENSITY ALTITUDE ~ 140 FT  
TRUE AIRSPEED ~ 15 KTS  
ROTOR SPEED ~ 354 RPM

GROSS WEIGHT ~ 2920 LB  
CG STATION ~ 407.4 IN (FWD)  
SKID HEIGHT ~ 12 FT



**FIGURE 77**  
**AUTOROTATIONAL ENTRY CHARACTERISTICS**  
**CH-47A USA S/N 68-16700**

CG STATION	AVG GROSS WEIGHT	DENSITY	ENGIN MOTOR	CONFIGURATION
-IM	-15	-1.1	SPEED RPM	
106.3 (FWO) 2728		3150	354	ARMED (DOORS ON)

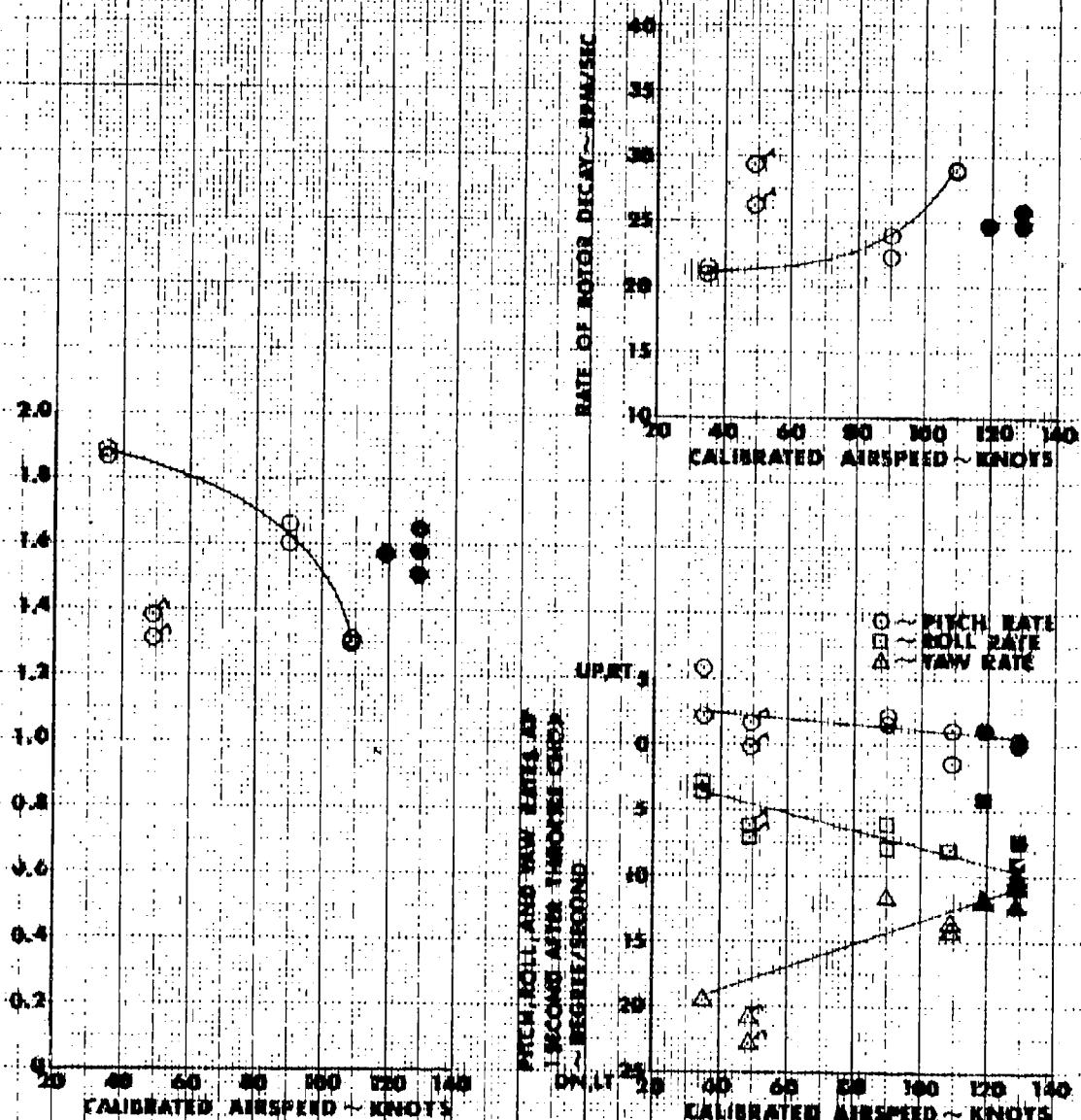
**NOTE:**

SHADeD SYMBOLS DENOTE MAXIMUM POWER DOWN

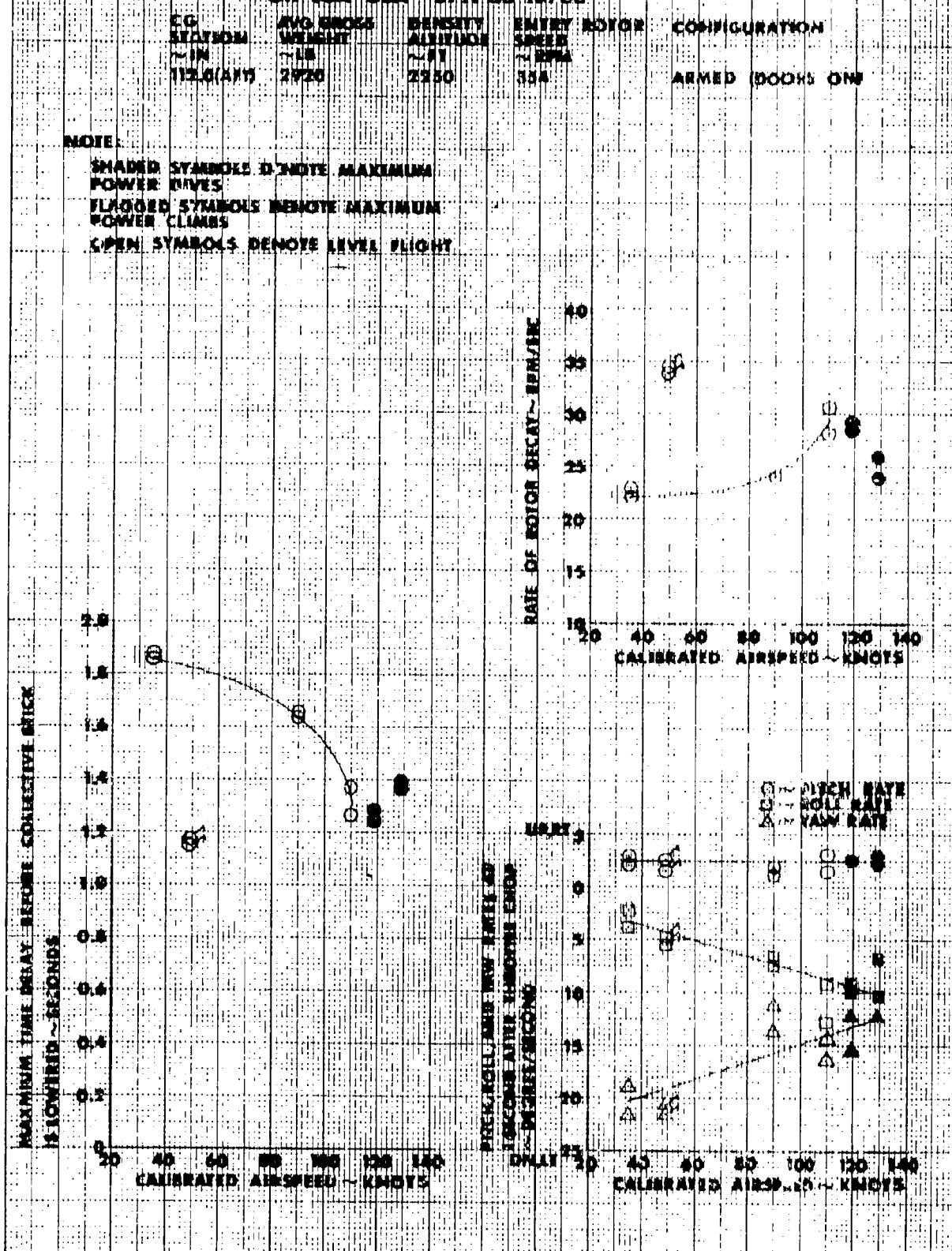
FLAGGED SYMBOLS DENOTE MAXIMUM POWER UPWARD

OPEN SYMBOLS DENOTE LEVEL FLIGHT

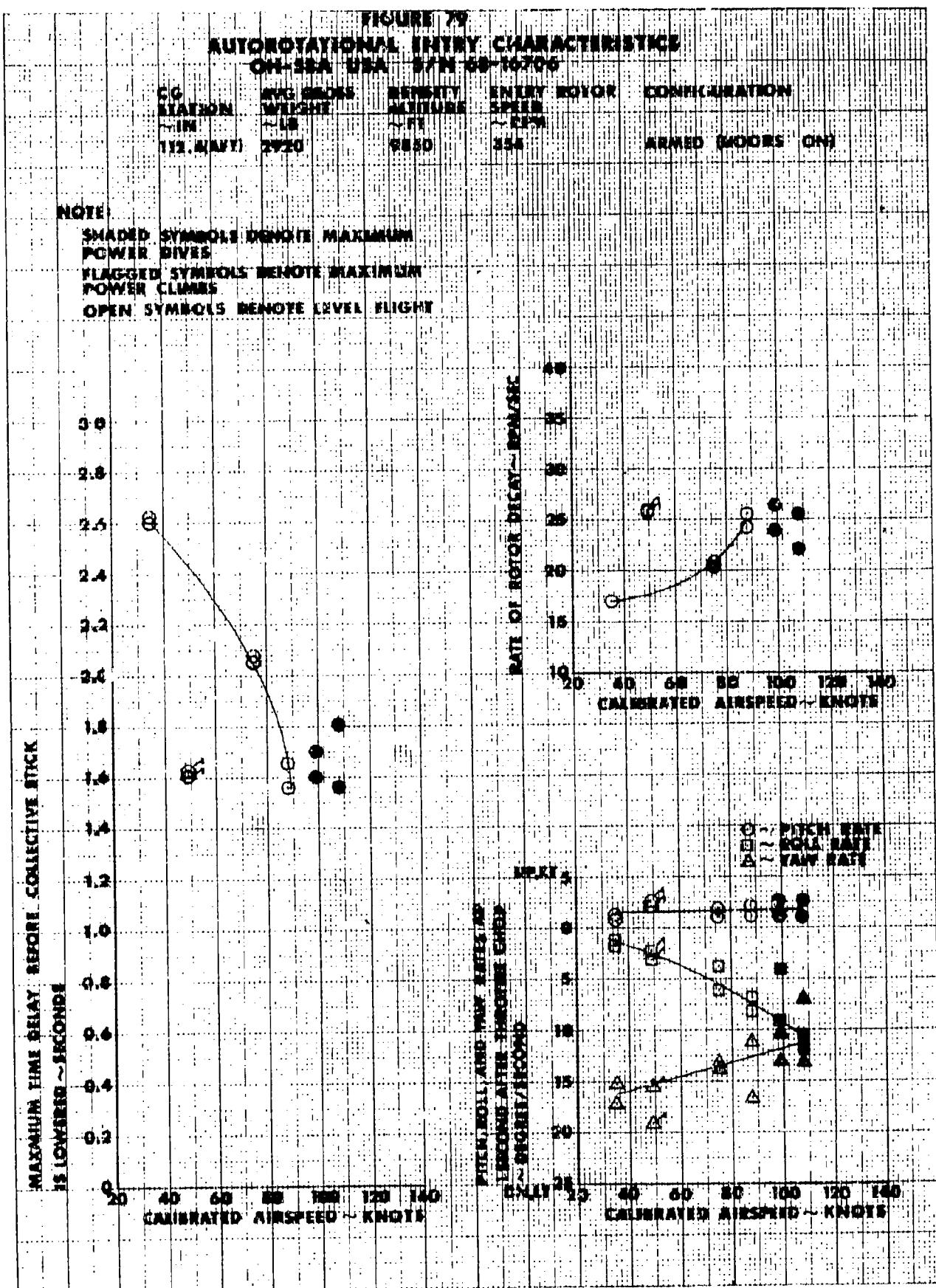
MATHTIM TIME BEHNEE CORRECTIVE STICK  
IS 1000000 ~ SECONDS



**FIGURE 7**  
**AUTOROTATIONAL ENTRY CHARACTERISTICS**  
**OM-SEA USA 8/19/68-1970**

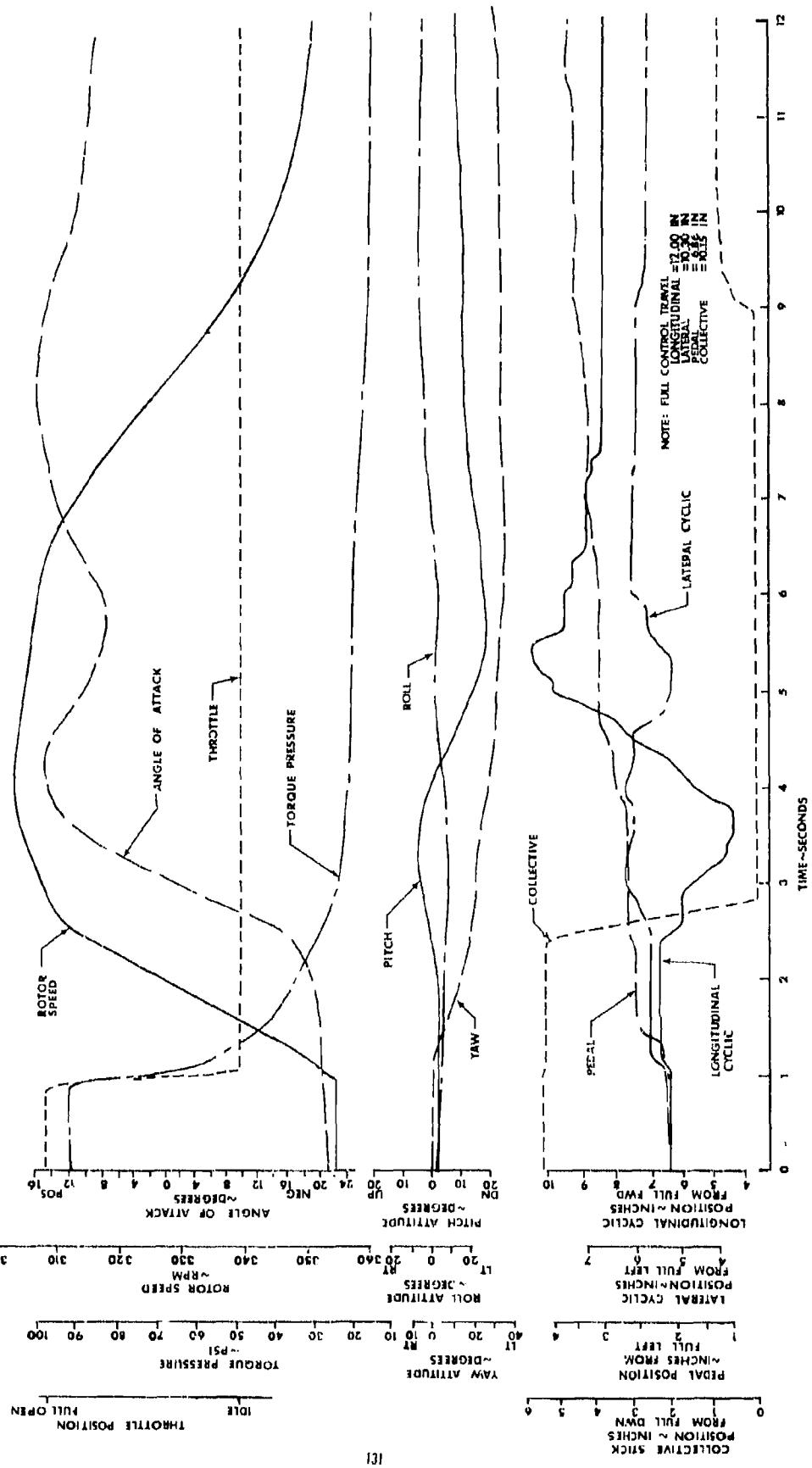


**FIGURE 79  
AUTOROTATIONAL ENTRY CHARACTERISTICS  
ON 5000' GROUND LEVEL - 00-102700**



**FIGURE 80**  
**AUTOROTATIONAL ENTRY FROM LEVEL FLIGHT**  
**OH-58A USA S/N 68-16706**

DENSITY ALTITUDE ~3100 FT	GROSS WEIGHT ~ 2690 LB
CALIBRATED AIRSPEED ~127 KTS	CG POSITION ~ 106.1 IN (FWC)
	CONFIGURATION ~ ARMED (DOORS ON)



**FIGURE 81**  
**CONTROLE PILOT PIKING WITH XM27E1 ARMAMENT KIT**  
**XM-27A HKA PN M-15206**

中華人民共和國農業部農業科學院植物保護研究所編《中國農業科學》

DEGREE	DENSITY	MOTOR SPEED
WAVELENGTH	~17	-RPM
2840	0130	354
2840	0130	354
2840	0130	354
2840	0130	354

	RATE OF FIRE ~ ROUNDS/MINUTE
CTION	4000
ZONTAL	2000
DOWN	4000
DOWN	2000
UP	2000

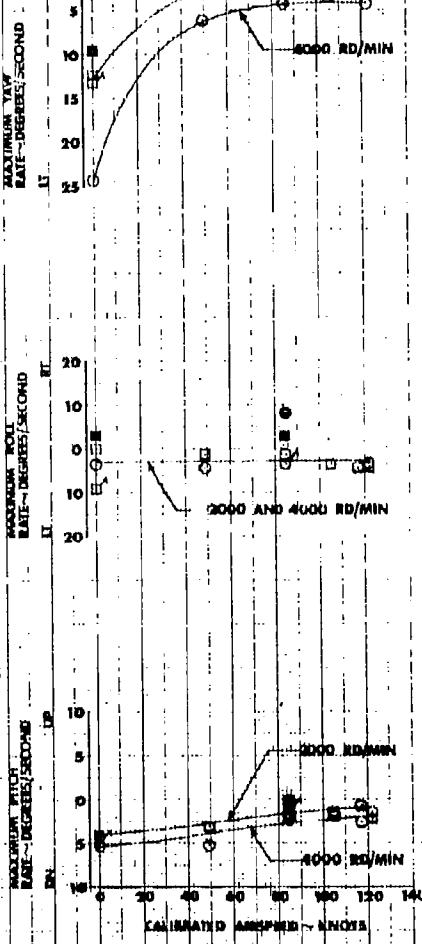
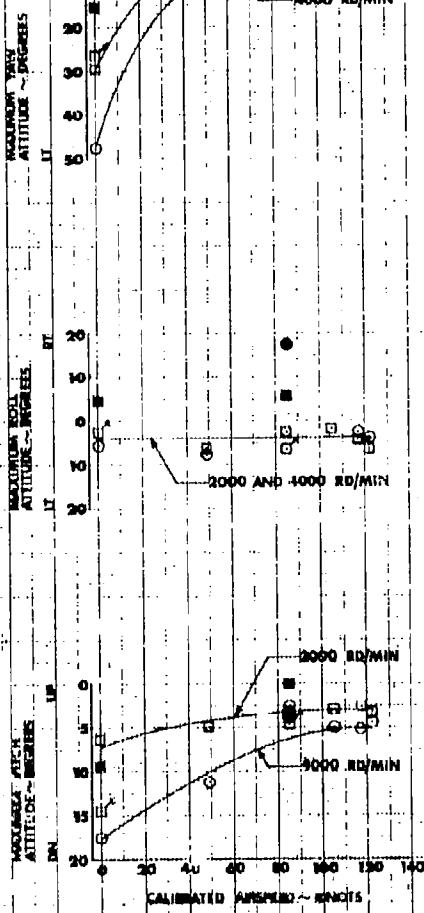


FIGURE 82

FIRING - LOW SPEED  
M-58 USA S/N 16706

04-38 USE 31 N 18/06

DENSITY ALTITUDE	~ 3470	LE	2880
COLORED AIRSPEED	~ 49	FT	IN [FWD]
BARO. SPEED	~ 49	KTS	107.8 IN [FWD]
	14.6 RDM		CG STATION
			CONFIGURATION

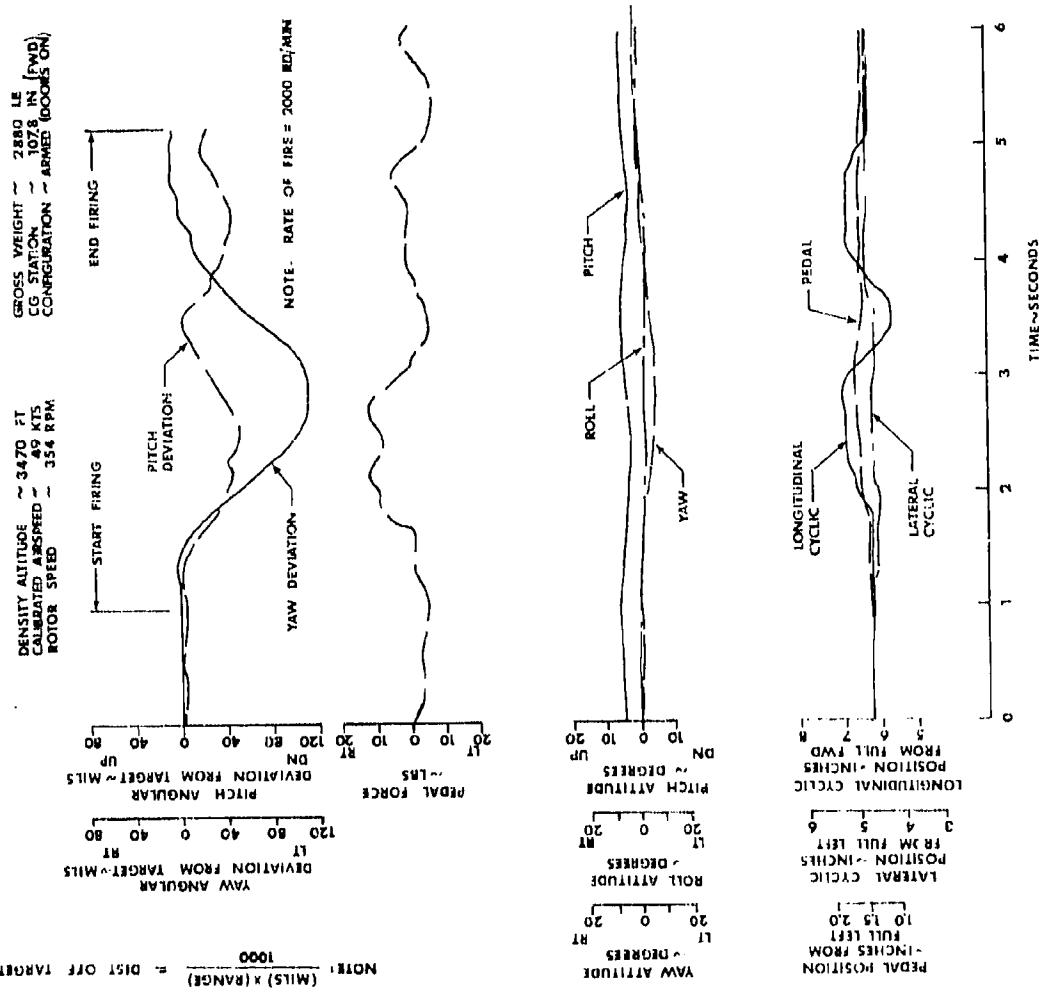


FIGURE 83  
FIRING - HIGH SPEED

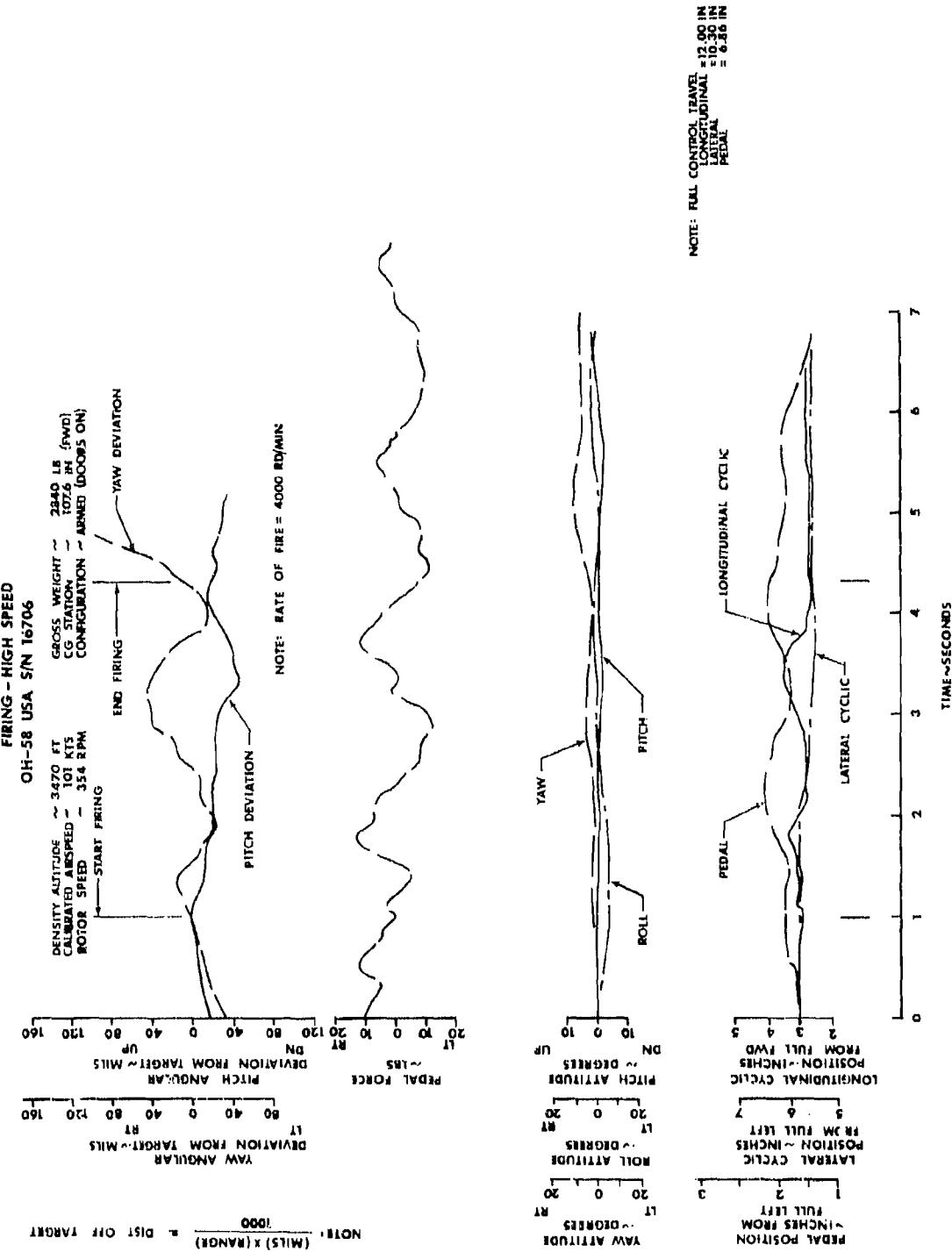
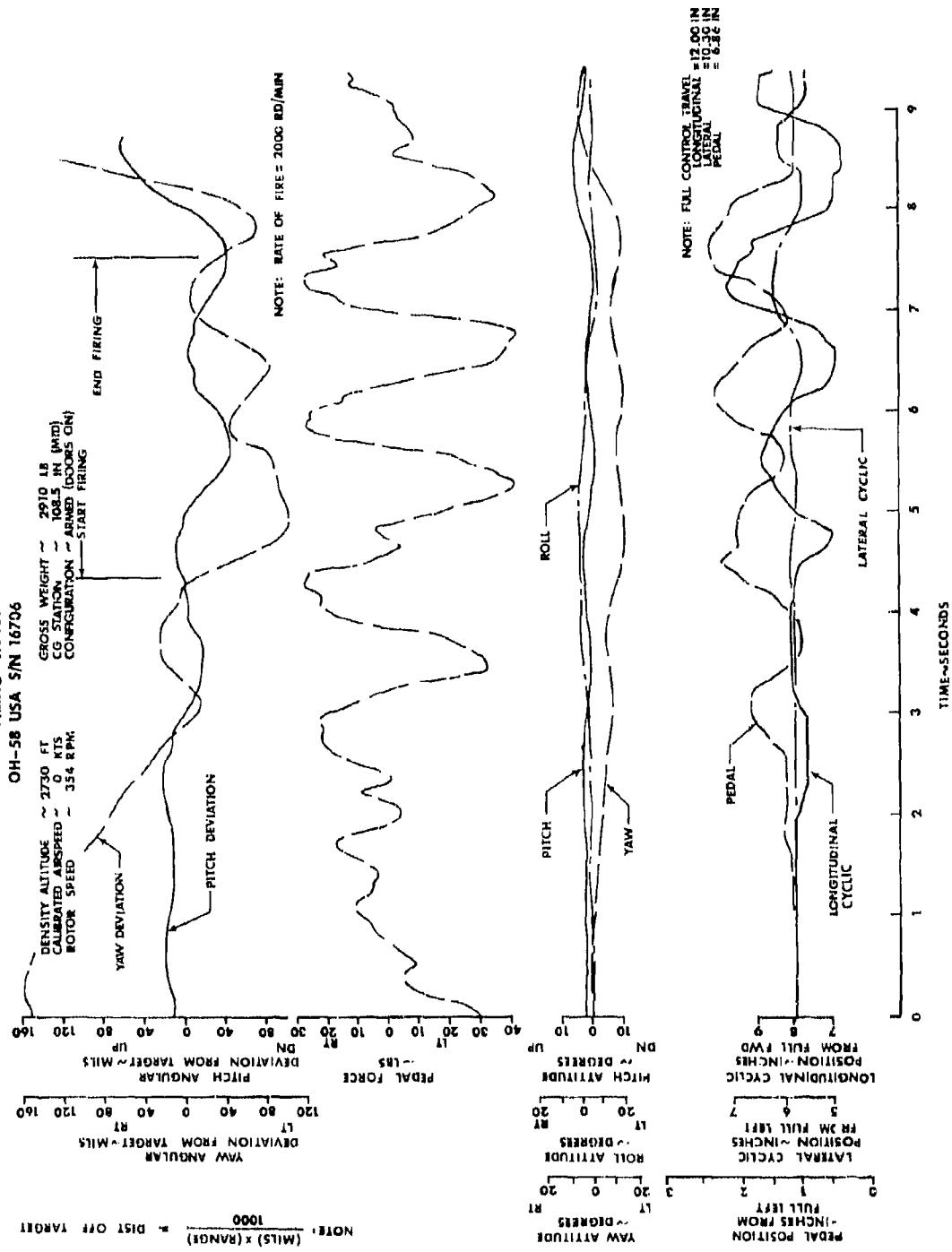


FIGURE 84

FIRING - HOVER



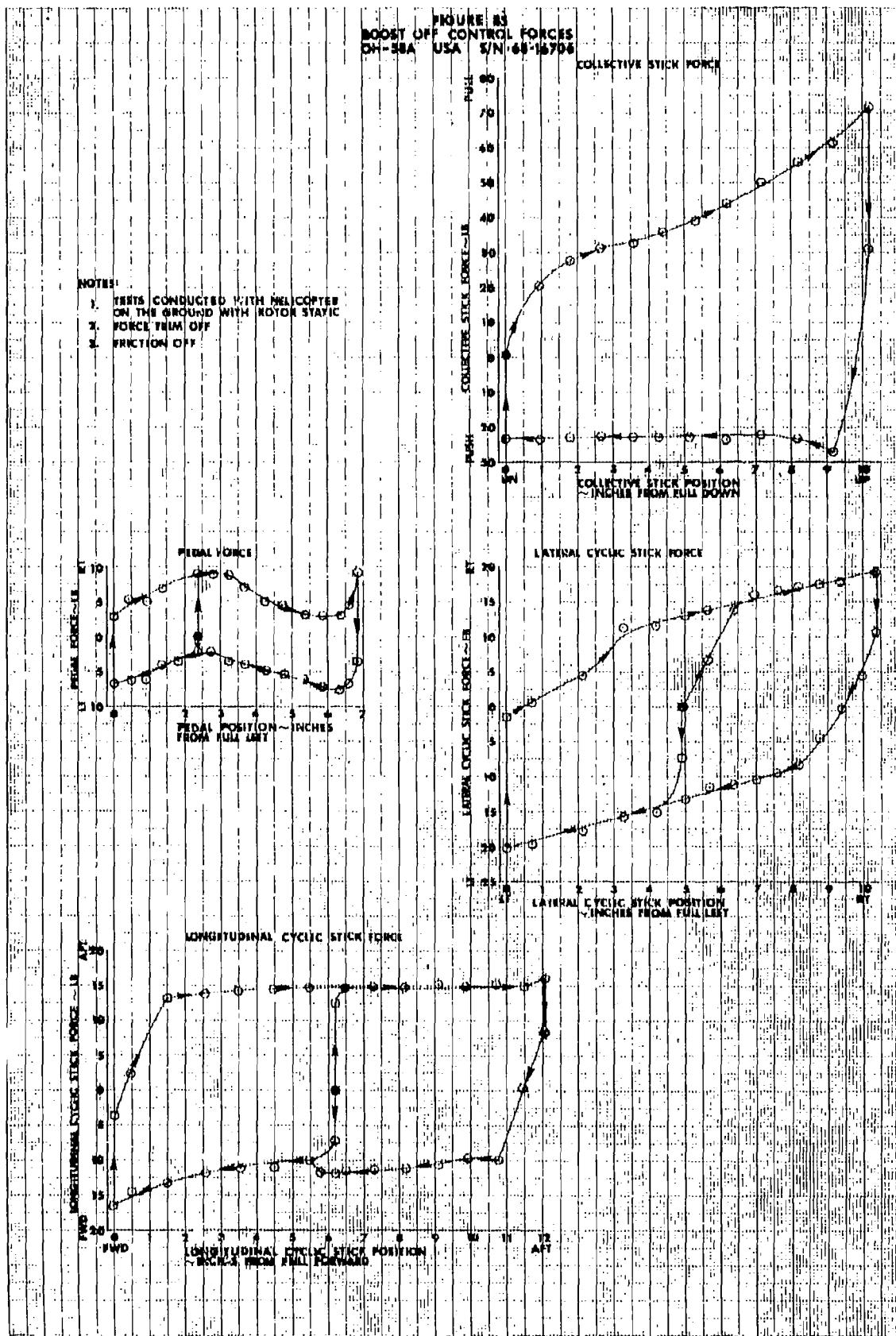
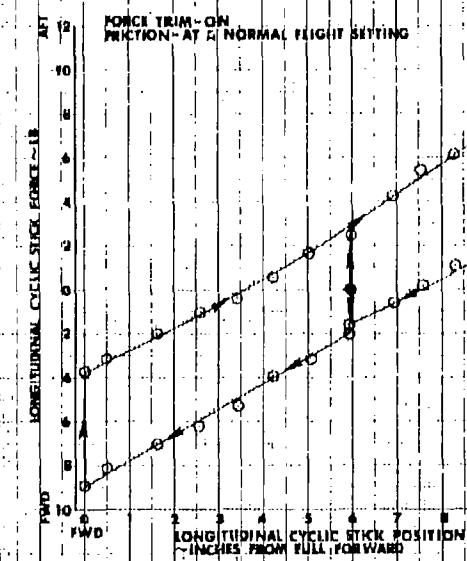
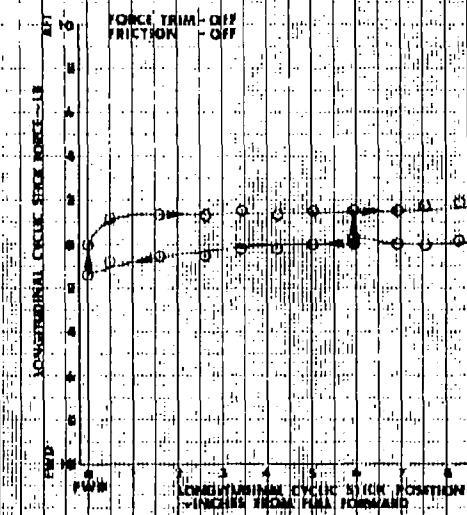


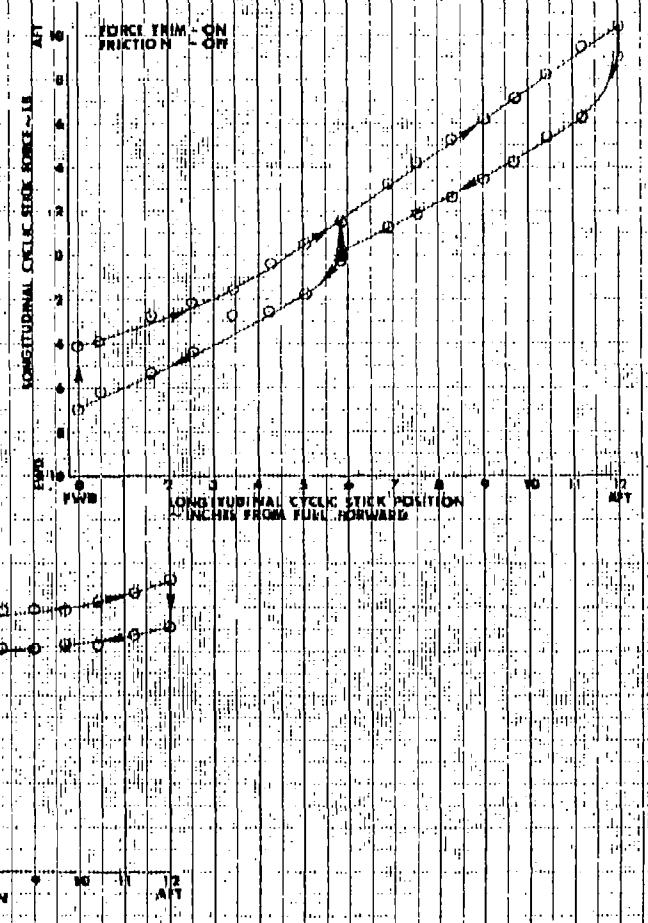
FIGURE 8A  
OH-58A USA S/N 68-16706



LONGITUDINAL CYCLIC STICK POSITION  
INCHES FROM FULL FORWARD



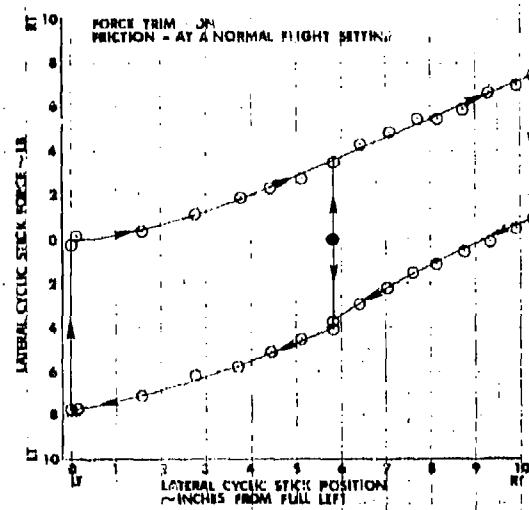
LONGITUDINAL CYCLIC STICK POSITION  
INCHES FROM FULL FORWARD



NOTES:

1. TESTS CONDUCTED WITH HELICOPTER ON THE GROUND WITH ROTOR STATIC.
2. LOADS ON TESTS CONDUCTED WITH HYDRAULIC PRESSURE SUPPLIED BY AN EXTERNAL SOURCE.

FIGURE 87  
LATERAL CYCLIC STICK FORCES  
OH-58A USA S/N 68-16706



NOTES:

1. TESTS CONDUCTED WITH HELICOPTER ON GROUND WITH ROTOR STATIC.
2. BOOST ON TESTS CONDUCTED WITH HYDRAULIC PRESSURE SUPPLIED BY AN EXTERNAL SOURCE.

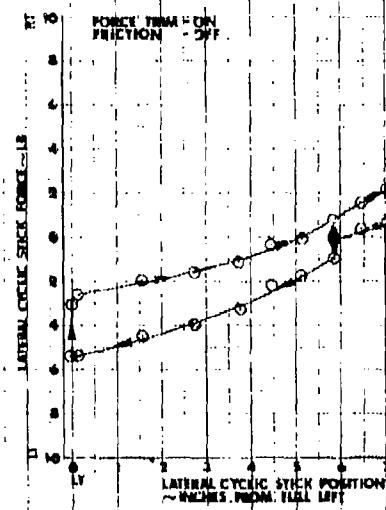
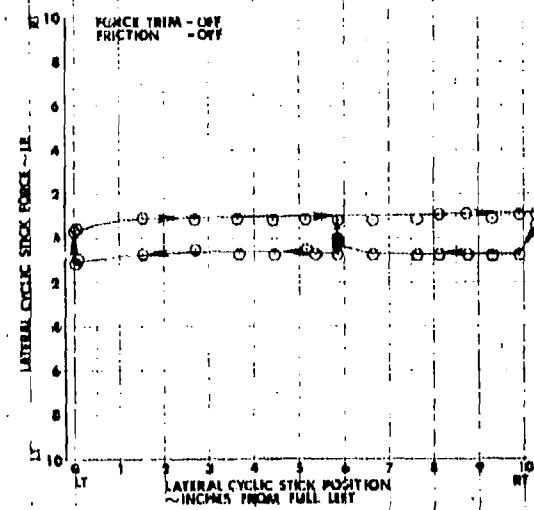
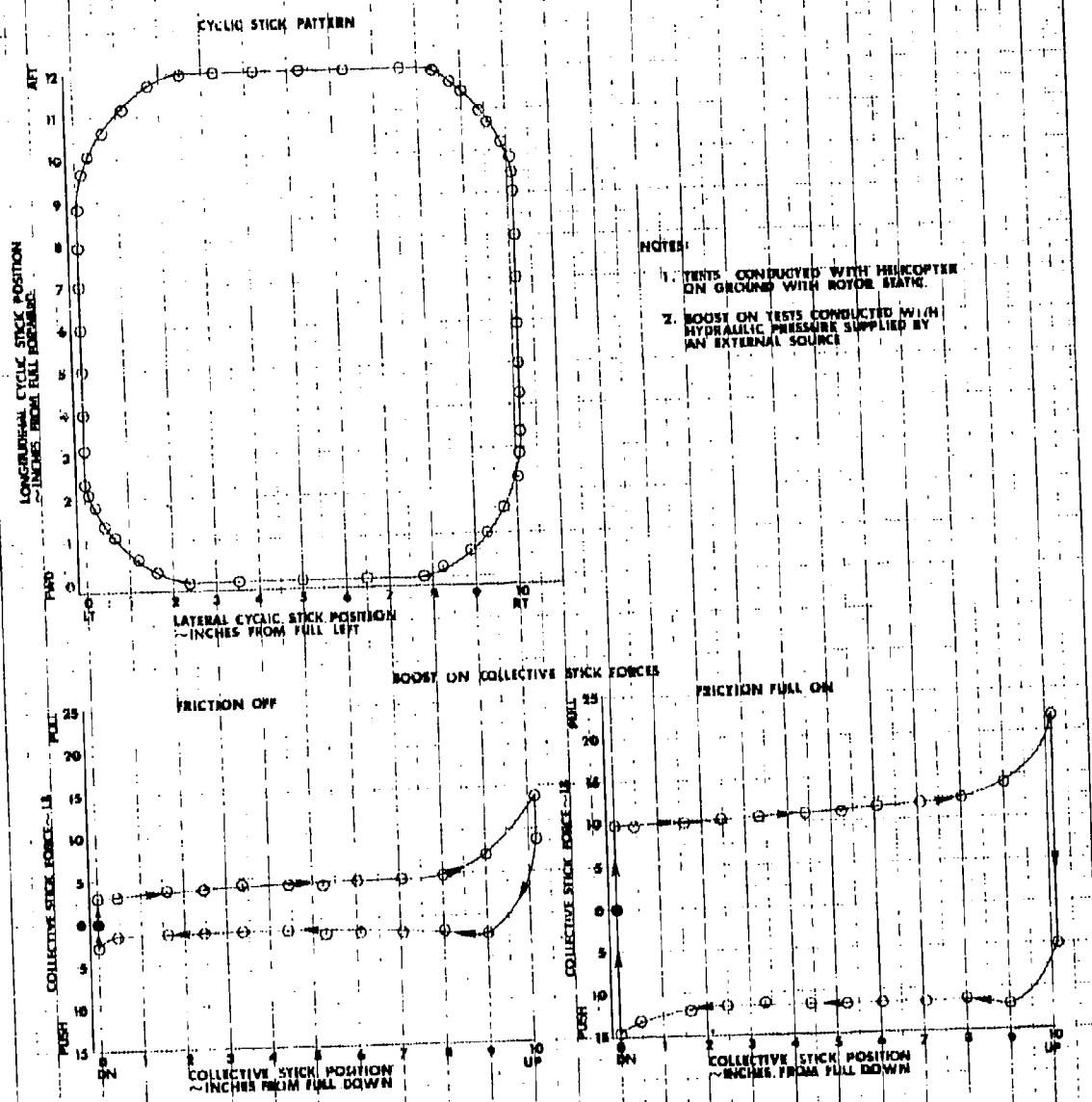
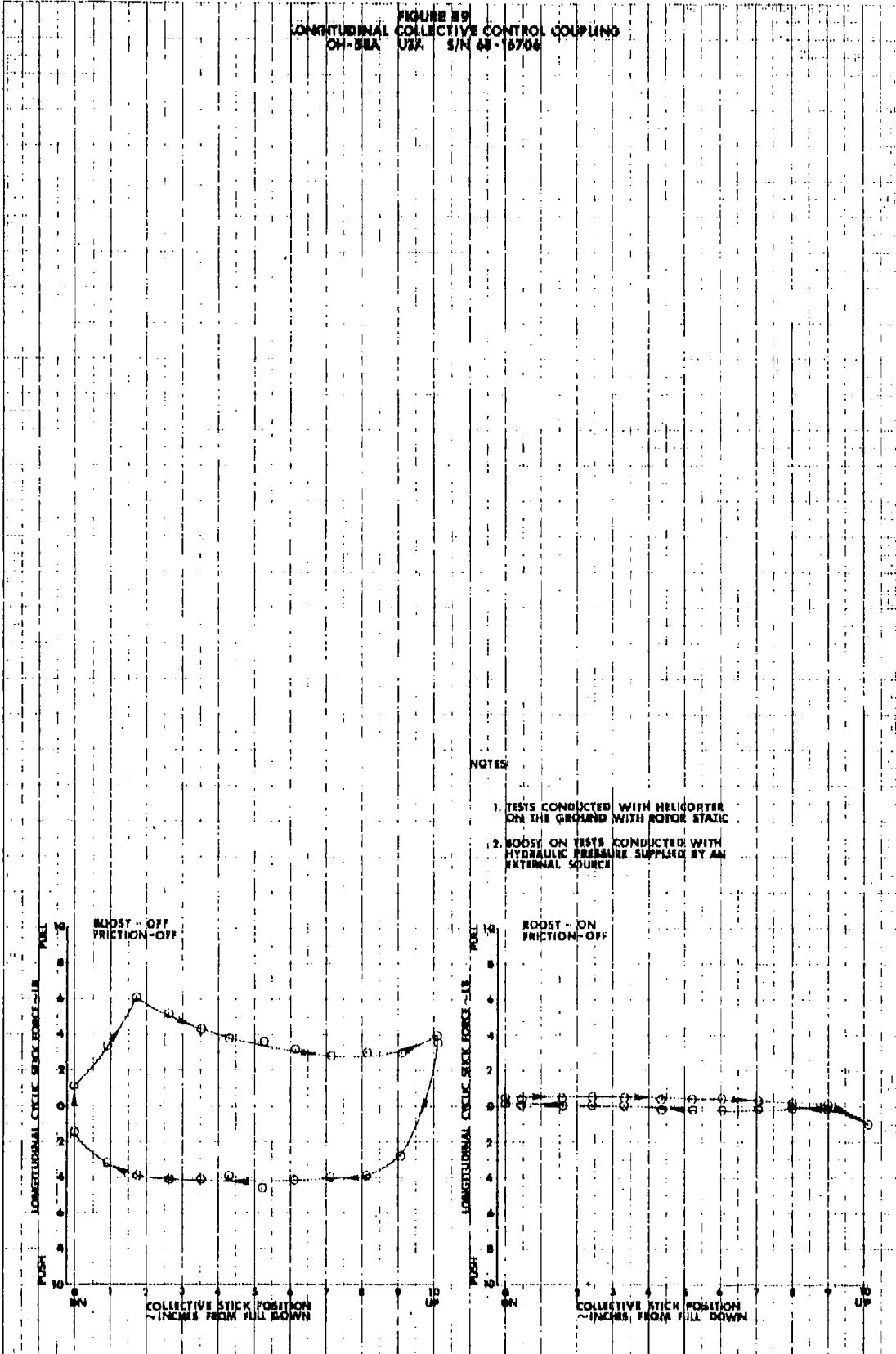


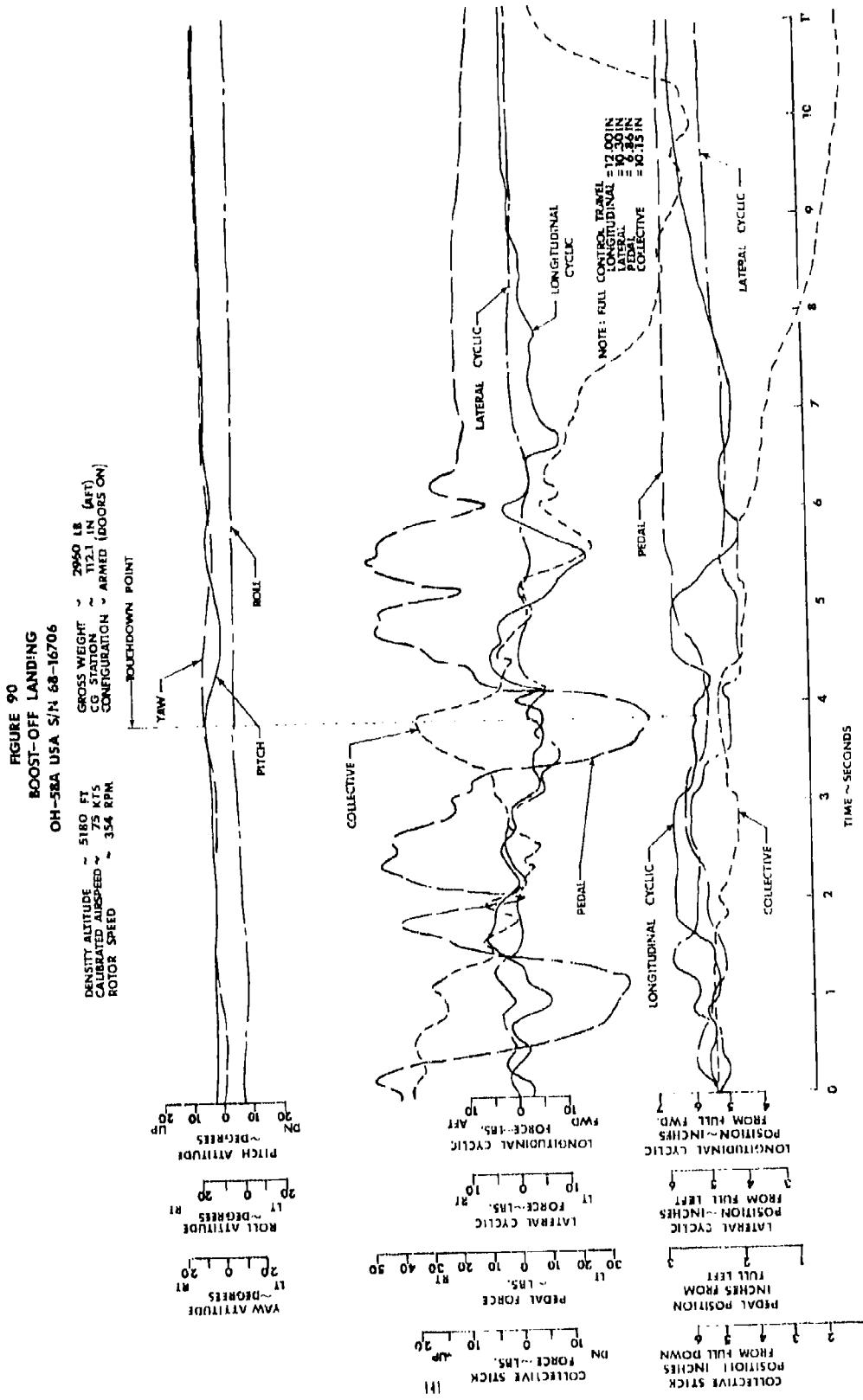
FIGURE 8A  
CONTROL SYSTEM EVALUATION  
CH-38A USA S/N 68-16706

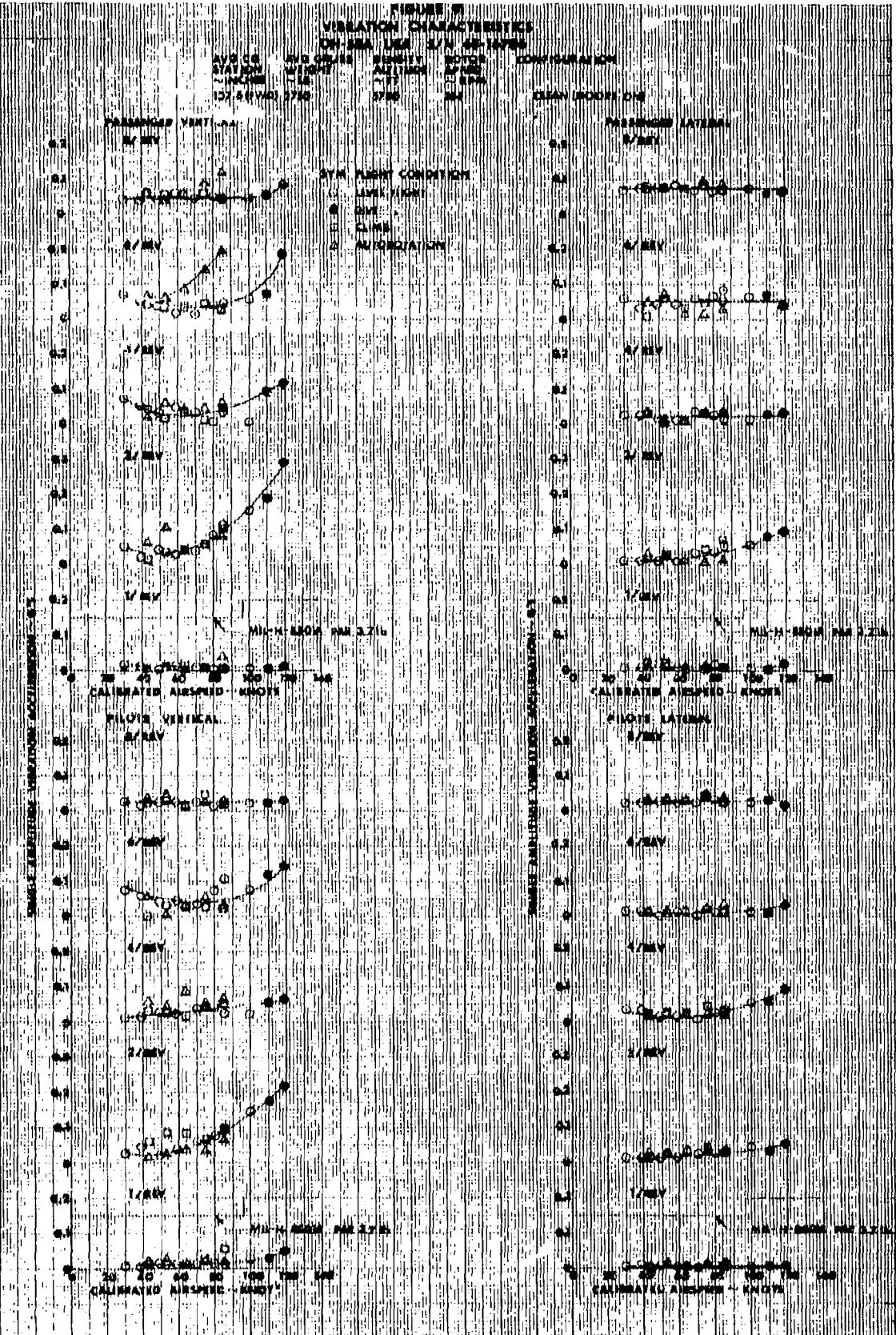


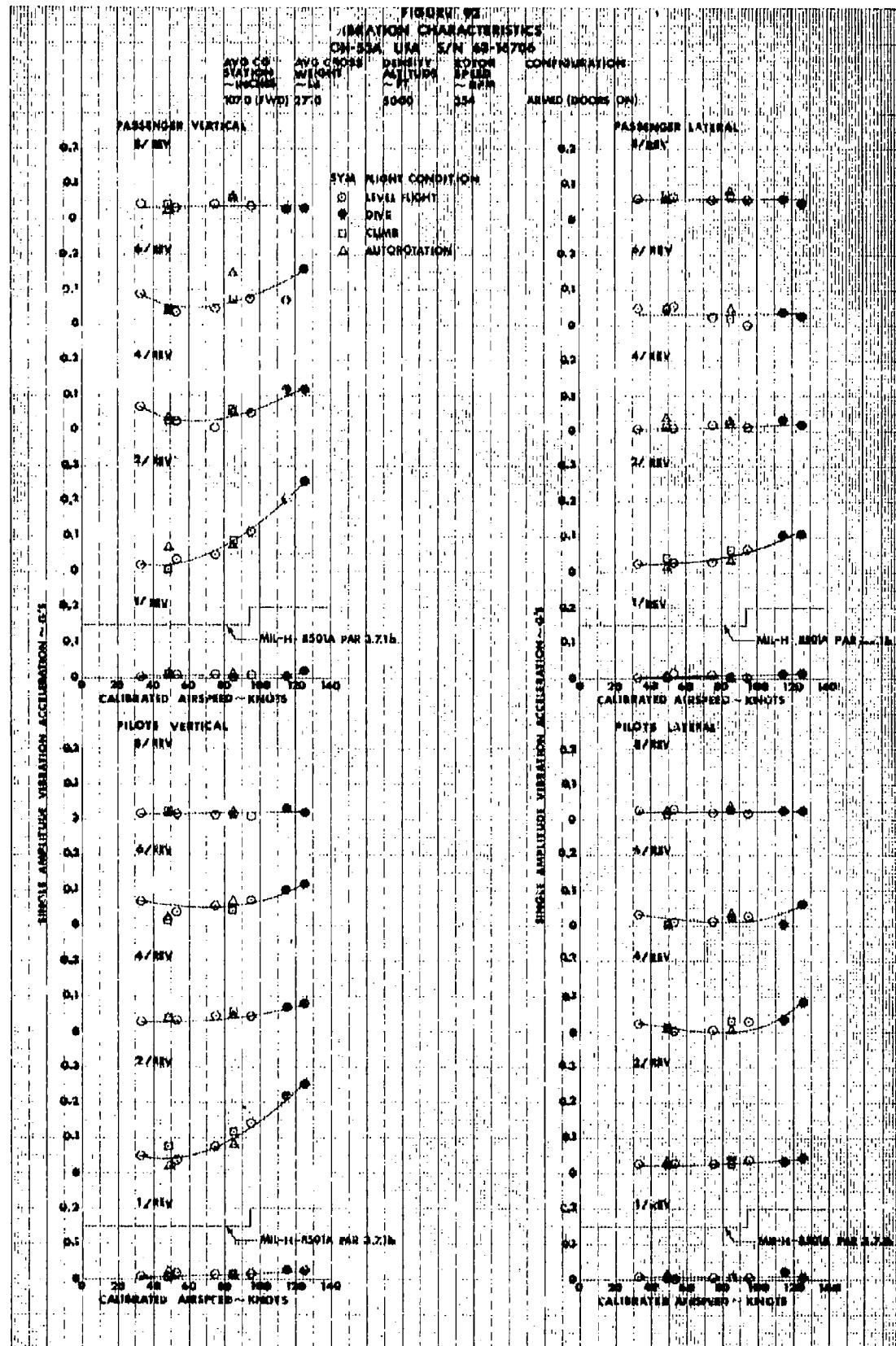
**FIGURE B9**  
**LONGITUDINAL COLLECTIVE CONTROL COUPLING**  
**OH-38A USA S/N 68-16704**

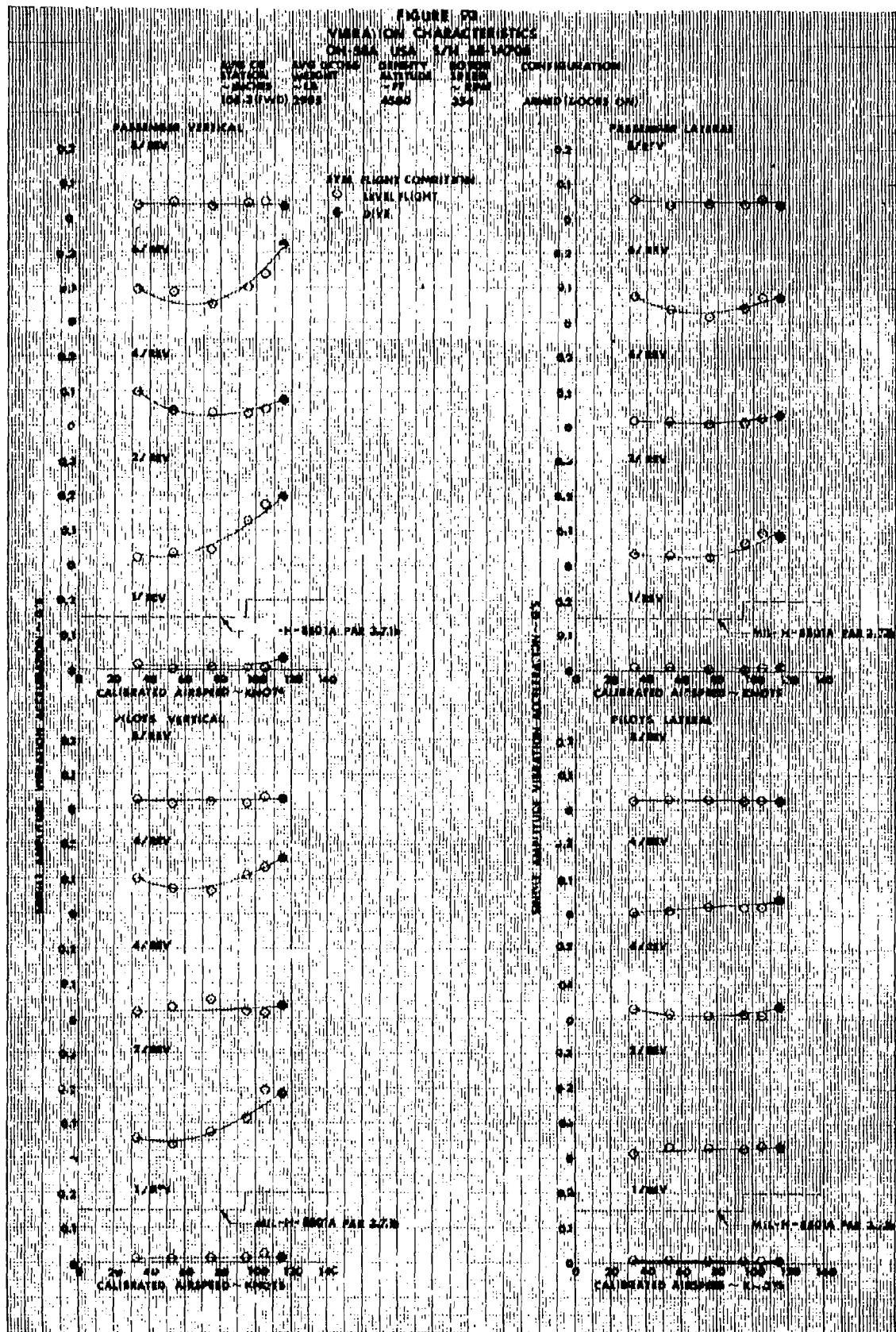


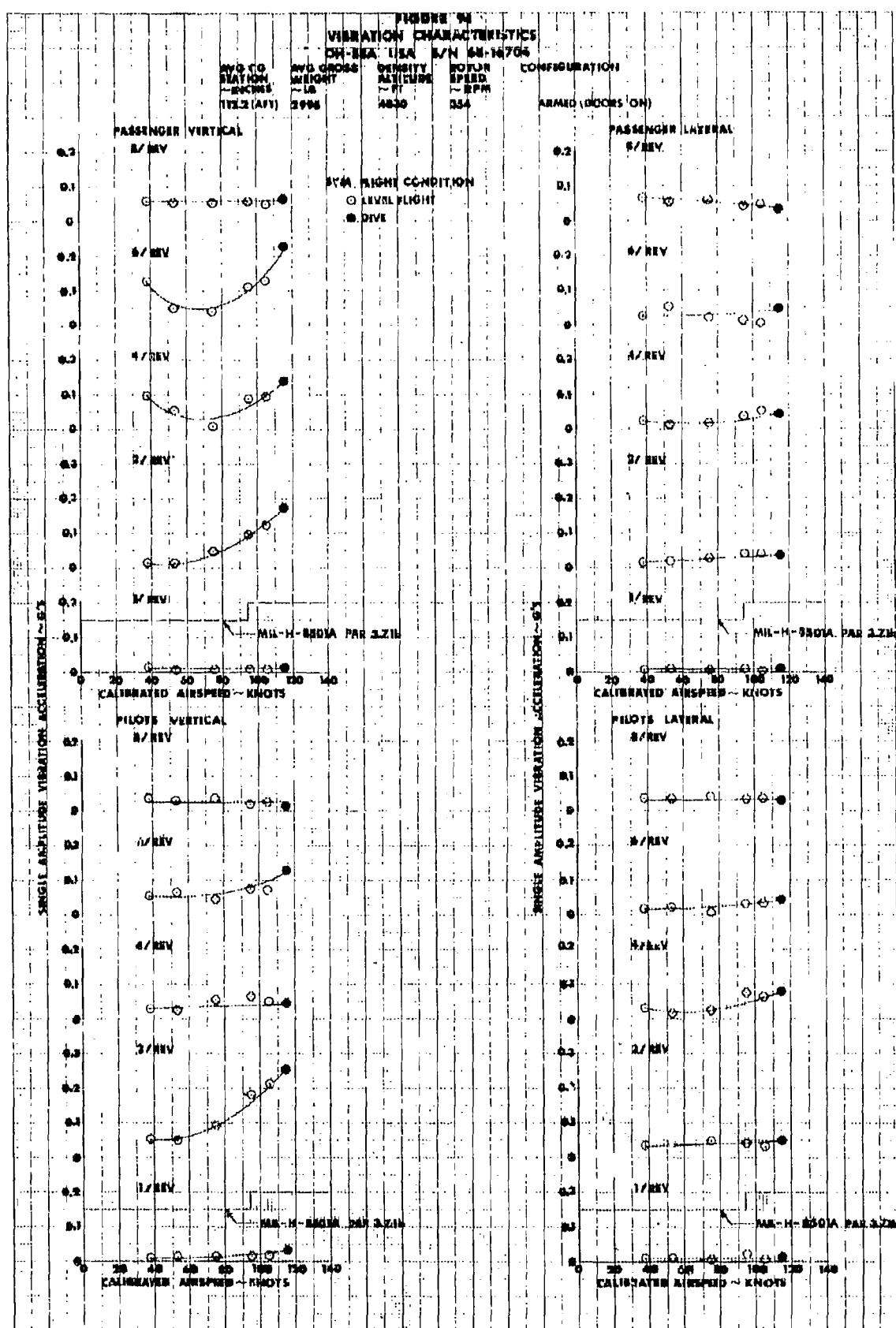
DENSITY ALTITUDE	~ 5100 FT	GROSS WEIGHT	2950 LB
CALIBRATED AIRSPEED	~ 775 KTS	CARGO STATION	112.1 IN (AFT)
ROTOR SPEED	~ 354 RPM	CONFIGURATION	ARMED (DOORS ON)

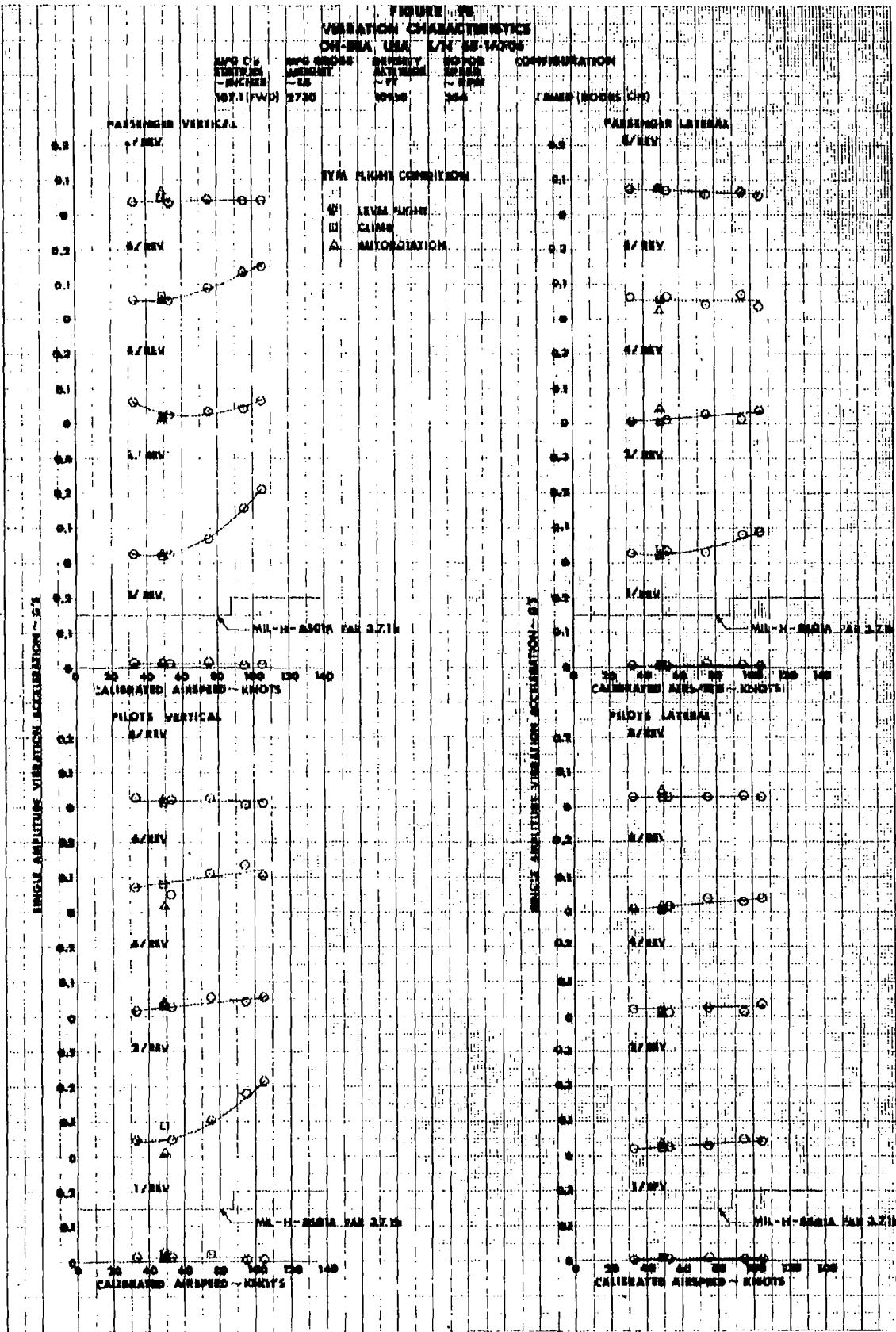












## APPENDIX V. SPECIFICATION NONCOMPLIANCE TABLE

Type of Test	Specification MIL-H-8501A Requirement	Specification MIL-H-8501A Paragraph	Results	Test Report Paragraph
Static longitudinal stability	3-inch maximum longitudinal trim change	3.2.10.2	Longitudinal cyclic trim change was 3.5 inches between climb and autorotation	21
Static lateral-directional stability	Pedal and lateral displacement gradients approximately linear between sideslip angles of ±15 degrees	3.3.9	Pedal and lateral gradients sometimes nonlinear	32
Static lateral-directional stability	Positive pedal and lateral control displacement gradients	3.3.9	Some negative gradients	32
Autorotational entry	2-second minimum delay before lowering collective	3.5.5	Delay times as low as 1.16 seconds in climb	62
Control system evaluation	1.5-pound maximum longitudinal breakout force	3.2.7	1.7-pound longitudinal breakout force	78
Control system evaluation	7-pound maximum pedal breakout force	3.3.13	9-pound pedal breakout force	80
Control system evaluation	Linear pedal force gradient	3.3.11	Nonlinear gradient	80
Control system evaluation	Pedal forces not excessive in steady sideward flight	Deviation 19 (detail spec)	Pedal forces considered excessive (49 pounds)	81
Control system evaluation	15-pound maximum pedal force in sideward flight	3.3.12	49-pound right pedal force, 38-pound left pedal force	81
Control system evaluation	3-pound maximum collective breakout force	3.4.2	4.3-pound collective breakout force	83
Control system evaluation	No control force coupling	3.4.3	0.5-pound force transmitted to cyclic by collective movement (boost ON)	84
Control system evaluation	0.2-inch maximum control free play	3.5.10	0.53-inch longitudinal control free play, 0.91-inch lateral free play (boost OFF)	86
Control system evaluation	1-pound maximum control force transmitted to cyclic by moving collective	3.4.3	6-pound force transmitted (boost OFF)	87
Control system evaluation	Maximum breakout forces: (1.5-lb long. and lat) (3-lb coll)	3.2.7, 3.3.13, 3.4.2	Breakout forces (boost OFF): (14.5-lb long.) (13-lb lat) (38-lb coll)	88
Vibrations	0.15-g maximum vibration level below cruise	3.7.1(b)	0.19g at 85 KCAS	90
Control response and sensitivity	Damping at least 4575 ft-lb/rad/sec	3.3.19	Damping approached zero	43
Control response and sensitivity	20-deg/sec/in. maximum lateral control response	3.3.15	24-deg/sec/in. lateral control response reached	41

## APPENDIX VI. DISTRIBUTION

<u>Agency</u>	<u>Test Plans</u>	<u>Interim Reports</u>	<u>Final Reports</u>
Commanding General US Army Aviation Systems Command ATTN: AMSAV-R-F AMSAV-C-A AMSAV-D-ZDOR AMSAV-D-W AMSAV-R-R PO Box 209 St. Louis, Missouri 63166	5 - - - - -	5 - - - - -	6 2 2 2 1
Commanding General US Army Materiel Command ATTN: AMCPM-LH PO Box 209 St. Louis, Missouri 63166	5	1	25
Commanding General US Army Materiel Command ATTN: AMCRD AMCAD-S AMCPP AMCMR AMCQA Washington, D. C. 20315	2 - - 2 -	1 - - - -	2 1 1 2 1
Commanding General US Army Combat Developments Command ATTN: USACDC LnO PO Box 209 St. Louis, Missouri 63166	11	11	11
Commanding General US Continental Army Command ATTN: DCSIT-SCH-PD Fort Monroe, Virginia 23351	-	-	1

<u>Agency</u>	<u>Test Plans</u>	<u>Interim Reports</u>	<u>Final Reports</u>
Commanding General US Army Test and Evaluation Command ATTN: AMSTE-BG USMC LnO	2 1	2 1	2 1
Aberdeen Proving Ground, Maryland 21005			
Commanding Officer US Army Aviation Materiel Laboratories ATTN: SAVFE-SO, M. Lee SAVFE-TD SAVFE-AM SAVFE-AV SAVFE-PP	- - - - -	- - - - -	1 2 1 1 1
Fort Eustis, Virginia 23604			
Commanding General US Army Aviation Center Fort Rucker, Alabama 36362	1	1	1
Commandant US Army Primary Helicopter School Fort Wolters, Texas 76067	1	1	1
President US Army Aviation Test Board Fort Rucker, Alabama 36362	1	1	1
Director US Army Board for Aviation Accident Research Fort Rucker, Alabama 36362	-	1	1
President US Army Maintenance Board Fort Knox, Kentucky 40121	-	-	1
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13. ABSTRACT  Stability and control tests were conducted on a production model OH-58A helicopter to evaluate its flying qualities in the unarmed configuration and in an armed configuration with the XM27E1 weapon system. Limited testing was also performed to evaluate the helicopter slope landing capabilities and flying qualities with skis installed. Human factors and maintainability characteristics were noted throughout the evaluation. Testing was performed by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California, between 6 October 1969 and 16 February 1970. The testing consisted of 89 flights which totaled 85.3 hours of productive flight testing. There were no deficiencies recorded. Eighteen shortcomings are reported. Difficulty in maintaining precise directional control during hovering flight is a shortcoming which warrants improvement on a priority basis. This shortcoming requires excessive pilot effort and degrades the accuracy in firing of the XM27E1 armament subsystem. It is recommended that a caution note be placed in the operator's manual warning against hovering in a tail wind in excess of 30 knots. The capability of landing on a 10-degree slope was marginal but is not considered to be a shortcoming. Flying qualities of the aircraft with skis installed were satisfactory. Maintainability of the helicopter was excellent throughout the test program.			

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